U. S. AIR FORCE

PROJECT RAND

RESEARCH MEMORANDUM

A FAMILY OF RECOVERABLE RECONNAISSANCE SATELLITES (U)

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SUMMARY

A family of reconnaissance satellites is proposed which would provide both early and continuing photographic reconnaissance capability in augmentation of the WS-117L program. The systems proposed differ substantially from the current 117L concept:

1. The proposed systems use a spin-stabilized payload stage.
2. They use a transverse panoramic camera of essentially conventional design, fixed to spin with the final stage, which scans across the line of flight.
3. The entire payload stage is recovered.

The first member of this family uses a 12-in. camera, carrying 500 feet of 5-in.-wide film. The extremely short exposure time--1/4000 sec--eliminates the need for precise altitude, exact image-speed synchronization, difficult vehicle-camera performance, and extensive monitoring and adjusting of the camera system. The system will provide sharp photographs of about 60-ft ground resolution. Each exposure, covering some 300 miles across the line of flight, will photograph some 18,000 sq mi. The 500-ft roll will cover some 4,000,000 sq mi (almost half of the U.S.S.R.) and show major targets, airfields, lines of communication, and urban and industrial areas. The satellite which should weigh about 300 lb could be placed in a polar orbit at 142 ± 47 miles altitude by a combination of rockets such as Thor, plus a second-stage Vanguard, plus a third-stage small solid-rocket similar to Vanguard's third stage. A one-day operation is envisaged, with recovery of the satellite by command firing of a braking rocket on the 16th pass, so that it would impact in a predictable ocean area.

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This, the first proposed system, is the easiest to build of the several possibilities; it could be followed by similar, but more sophisticated systems using longer focal-length cameras and larger film loads. These advanced systems would provide much more detailed reconnaissance over larger areas than the early system. Availability of the first system is believed to be about one year from the date of contract.

The 'later' systems could utilize 36 in. and 120 in. lenses, as described in this report. Based on experience with many other Research and Development projects, it is suggested that these systems be built in this order, for attempts to build the most sophisticated system would require building the simpler systems anyway and labeling them scale test models.

PREFACE

This research memorandum is the product of a feasibility study undertaken by several members of the RAND technical staff. The basic idea of using a spin-stabilized panoramic camera in the manner described herein was developed by M. E. Davies in the spring of 1957. Others who contributed in their special technical fields are listed in the table of contents.

The study presents a concept of photographic reconnaissance for which satellites appear eminently suited, and a preliminary design of the first member of this reconnaissance satellite system, based on hardware now available or which could be available in the near future. The choice of specific design parameters and methods of operation must be determined on the basis of further engineering studies. There is nothing unique or limiting in the choices of lens focal lengths or film payloads.

For the reason noted in the Summary (p. iv), design of the more complicated systems has been largely ignored in this report. However, optical and camera designs are available to meet the essentially conservative specifications. Detailed design of advanced versions of the type of satellite proposed must be postponed until certain concepts have been tested with simpler systems.

For example, for purposes of proposal and calculation, an interesting alternate to the first system (the 12-in. focal length--40 lines/mm--1/4000th sec--Plus X Aerecon combination) would start with an existing Baker 24-in. f/5.0 telephoto lens, easily capable of 100 lines/mm on a microfile emulsion, with exposures at 1/1000 sec. These numbers, and this system, would yield a ground resolution five times better than the preceding 12-in. system--a ground resolution of 12 feet! Exercise of this option, however, and exploration of this possibility with its tighter tolerances, should follow, not precede, testing of the simplest possible system. For this reason, no further mention of this otherwise intriguing possibility is made in this report. There are other and important aspects of any reconnaissance system which are not treated in this report, e.g., the required accompanying ground handling system, the way in which measurements can be made from panoramic photography, and related questions. These are all legitimate topics for subsequent investigation and detailed discussion.

Following many paragraphs in the body of the text, there are references to various appendixes. These are intended to clarify or develop specific remarks. Each appendix has, for editorial purposes, been treated separately; bibliographical references and other supporting data are therefore given at the end of each appendix.

CONTENTS

I. INTRODUCTION

This memorandum describes a reconnaissance satellite system that would provide an early and continuing photographic reconnaissance capability in augmentation of the WS-117L program. Relatively simple in operation, the system would use a camera of essentially conventional design in a comparatively unsophisticated orbiting vehicle. A launching date about one year from the date of contract is contemplated. The system will produce pictures of a scale and resolution that will yield valuable intelligence information about large areas of the Soviet Union.

Sections II through VII below discuss the reconnaissance satellite mission, the camera proposed for the system, the means for placing it in orbit and recovering the film, the intelligence payoff, and finally, the growth potential of this system considered as the first of a series of similar, but more advanced, reconnaissance satellites.

II. RECONNAISSANCE: NEEDS AND MEANS

It is acknowledged that there is a need for better military intelligence on the USSR and that aerial photography is a preferred means of collection. For one thing, the area occupied by the Soviet Union and its political satellites is very large and, for the most part, inaccessible except by overflight. Secondly, in the immediate future it will be vital for us to know a great deal about the patterns of use, installation, and concealment of Soviet ICBM's. Finally, it is essential that we have detailed information from time to time on aircraft-missile phasing in the Soviet Union. We must know the character and composition of these major threats to our lives and security.

In describing airborne photographic reconnaissance systems, it is convenient, by way of developing an operational concept, to think in terms of four levels of reconnaissance: A, B, C, and D.

Level A provides large-area search, measured in millions of square miles. Level B is limited-area search, measured in hundreds of thousands of square miles. Level C, specific-point-objective photography, is measured in hundreds of square miles. And Level D, technical-intelligence-objective photography, provides coverage in blocks tens of square miles or less in size.

The reconnaissance satellite system proposed permits us to progress systematically from Level A toward Level D in a series of system improvements. The basic systems will enable us to cover millions of square miles of the Soviet Union giving us photographs of such a scale and resolution that significant intelligence information can be obtained. Such missions can be repeated from time to time to reveal new developments in the

Soviet posture. Reconnaissance at Level A will also be valuable in providing information on where to conduct more detailed reconnaissance. While the system will not provide us with warning intelligence, it will help us estimate Soviet capabilities and identify certain kinds of major targets. (Appendix A)

III. THE CAMERA

The camera proposed for this system is a transverse panoramic camera containing a 12 in. focal-length, highly corrected f/3.5 lens which covers a fairly narrow angle of approximately 21 degrees. Wide-angle scanning is accomplished by the expedient of moving the lens across the field during the exposure time.

For transverse scanning of the ground from a satellite, the camera must rotate around the longitudinal axis of the vehicle. For this application it is proposed to rotate the entire orbiting vehicle with the camera firmly attached, thus generating a sweep across the line of flight. It is not proposed to rotate the camera within its carrier.

Figures 1 and 2 show the geometry of the camera, lens, and focal plane relationships. The lens is mounted perpendicular to the carrier's roll axis behind a quartz window in the surface of the carrier. The lens images the ground on a fixed slit in front of the focal plane, the slit serving, in effect, as a very fast shutter. When the film is moved during the scan exposure, at a rate exactly matching the image motion produced by passage of the carrier over the ground, a continous, sharp photograph is produced in the focal plane. During the portion of the rotational period in which the lens does not "see" the ground, a measured length of film is rolled off the supply spool in readiness for the next exposure. At the same time, the last exposure is wound up on the take-up spool. (Appendix B)

ATTITUDE STABILIZATION

Note that the transverse panoramic camera under discussion does not require the usual kind of attitude stabilization necessary for cameras

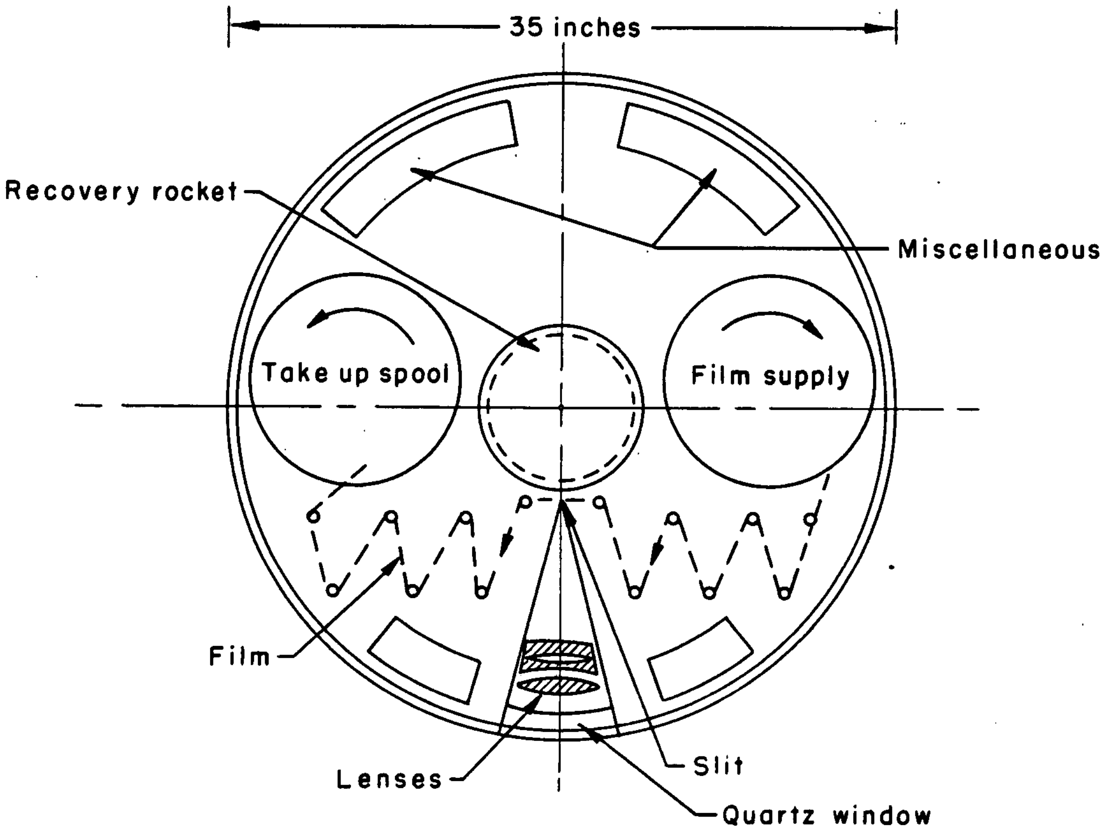


Fig. 1 — Camera geometry
Center section

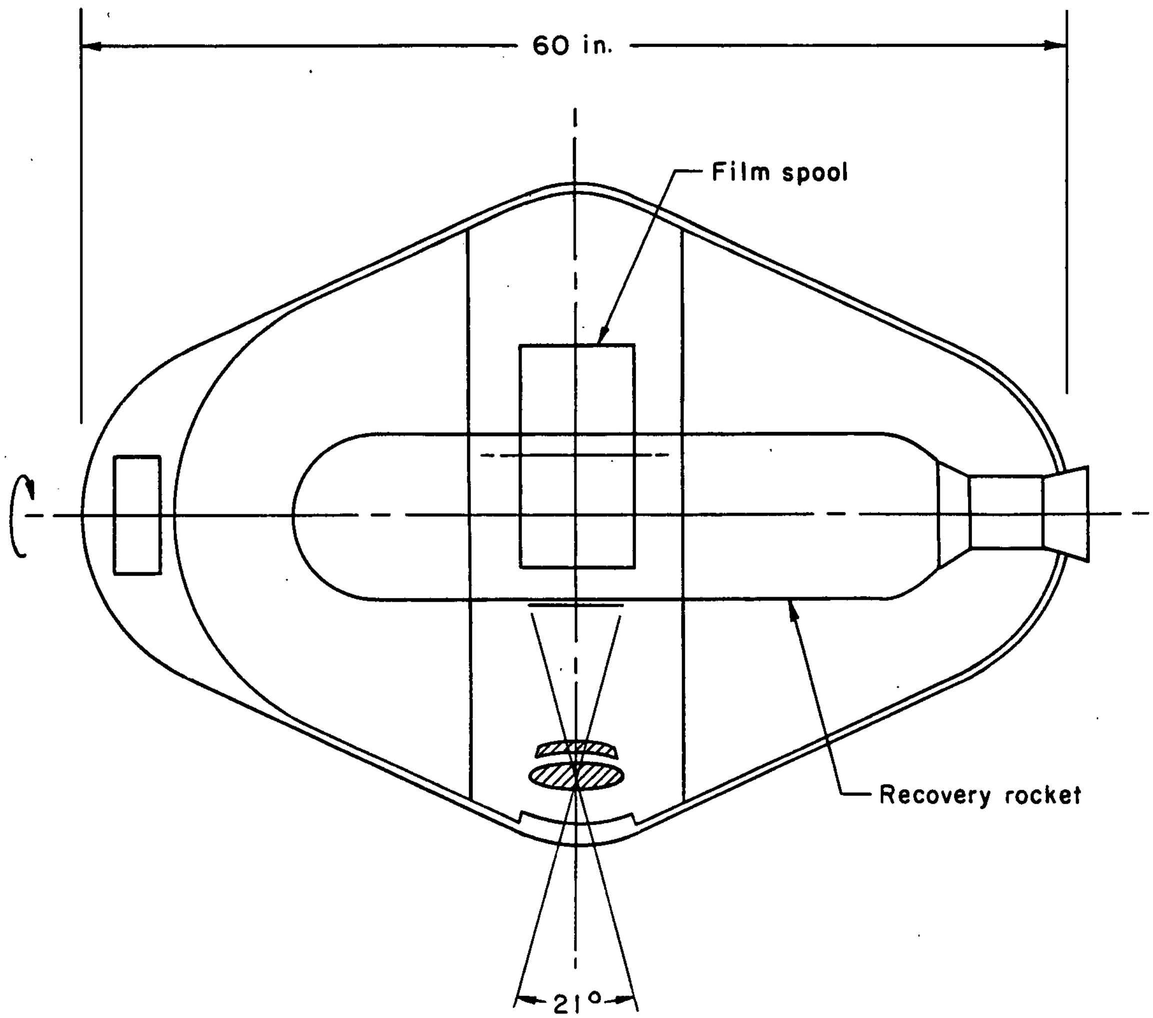


Fig. 2 — Camera geometry
Side view

mounted in aircraft. The entire carrier rotates; this is an important distinction between this proposal and other proposals for camera-carrying satellites.

Spin is imparted to the payload stage to produce the scan necessary for producing photographs across the line of flight. In addition, spin imparted to the payload stage stabilizes it in inertial space. That is, spinning the payload stage serves the twofold purpose of stabilizing the attitude of the camera in space and scanning the ground at the proper rate.

The preferred orientation of the camera in space is determined by the geography of the area to be surveyed (see Fig. 3). Although the Soviet bloc occupies about 180° of longitude, its area is somewhat compressed in latitude. Most of the territory of interest, from a reconnaissance standpoint, lies between 40 degrees North and 70 degrees North. This geographical fact turns out to be a very fortunate one as regards attitude stabilization, for it means that if the payload vehicle can be stabilized in an attitude horizontal to the surface of the earth at 55 degrees--i.e., midway between the two latitudes--the camera will produce acceptable pictures over the entire distance from 40 degrees North to 70 degrees North. (The satellite is considered to be on a polar orbit, as will be discussed below.) Note from Fig. 3 that, when the payload stage is at 40 degrees North, the vertical, with respect to the axis of the vehicle, is pointing back 15 degrees with respect to the earth; at 70 degrees North, it points forward 15 degrees. These angles result in very small errors in uncompensated image speed; in fact, they can be completely disregarded when discussing the quality of the photography. (Appendixes B and C)

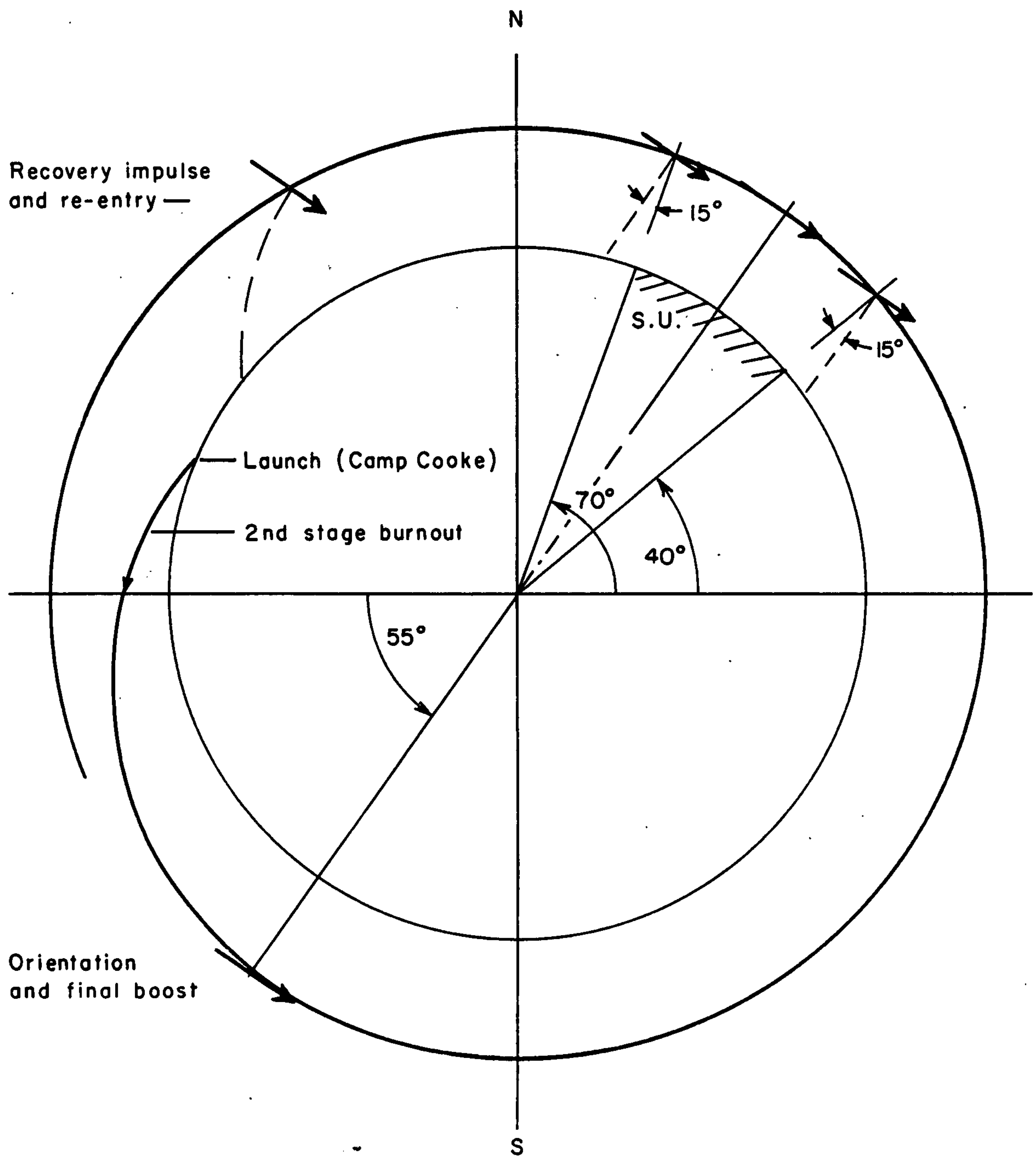


Fig. 3 — Schematic of trajectory and payload attitude

SHUTTER SPEED

Another important feature of the camera lies in the employment of a very high effective shutter speed, or short exposure time. This allows us to ignore fairly substantial changes in altitude and uncompensated image speed, changes in vehicle velocity over the surface of the earth, small angular rates and displacements, and other such effects which, in any customary reconnaissance vehicle, would certainly ruin photographic quality.

It appears, at first sight, that taking photographs from a satellite moving at about 18,000 miles per hour would be an extraordinarily difficult job because of image blurring, the lack of sharpness, the lack of definition, and hence the lack of information-gathering capacity. This first impression is not entirely erroneous. It is difficult to take pictures from an object moving at high speed--but not impossible. It can be done if the image motion during exposure is kept as small as possible, consistent with the requirements for definition.

This satellite reconnaissance system is not intended to achieve microscopic resolution at, say, levels of 100 lines/mm. The goal, believed to be fairly easily attainable, is a modest 40 lines/mm of film resolution. This is attained operationally with certain specialized reconnaissance systems now in use.

A statistic commonly used in describing aerial reconnaissance systems is ground resolution. Ground resolution is simply the ground dimension that corresponds to one line of resolution in a focal plane. In this case a 12-in. focal-length lens and 40 lines/mm are being considered. Thus 1/40 mm projected through a 12-in. lens to the ground at a distance of 750,000 ft corresponds to a distance of 60 ft.

The effective design shutter speed for this camera is 1/4000 second. This speed is obtained by properly designing the slit described earlier, in conjunction with the film speed and rate of scan, or vehicle rotation. This exposure speed is consistent with the choice of film emulsion--called Plus-X Aerecon--and the choice of lens speed f/3.5. At the design altitude, about 6 ft of forward motion are produced during the exposure time of 1/4000 sec while the vehicle is moving at 25,000 ft/sec. This is 1/10 of the basic 60-ft resolution element, and can be tolerated, in fact ignored, for purposes of photo interpretation.

This extremely fast exposure time makes it possible to ignore forward image motion; it also makes it possible to ignore altitude changes, probable angular displacements, velocity variations, and substantial variations in film rate. (Appendix B)

IV. THE ORBIT

A polar orbit is preferred, with a firing south from Camp Cooke. A design altitude acceptable to the camera is 142 $\pm$ 47 miles. If the satellite went much above 200 miles, ground resolution would be degraded. If altitudes much lower than 100 miles were used, the satellite would encounter undesirable aerodynamic forces. (Appendix D)

GROUND COVERAGE

With respect to total time in orbit, a 1-day operation is envisaged. In one day the satellite will make about 16 revolutions around the earth, six or seven of which would occur over the Soviet Union. The camera carries 500 feet of 5-inch-wide film which will permit 300 exposures at a film speed of about 23 inches per second. The payload rotates at about 20 revolutions per minute, with the camera timed to make an exposure every third revolution.

Each exposure will produce a picture covering some 18,000 square miles on the ground. Each pass over the Soviet Union will cover about 3/4 million square miles on the ground. The total film load will photograph some 4 million square miles, or nearly half the total land area of the Soviet Union. Fig. 4 is a schematic representation of a 1-day operation. (Appendix B)

ASCENT TRAJECTORY

The powered-ascent trajectory is effected by the combination of the Thor booster, with first-stage guidance preprogrammed for the autopilot, and the second-stage Vanguard using its autopilot in conjunction with components of the G.E. radio system.

During the period of coast to the design altitude of about 142 miles, the second stage, containing the spin-up and separation mechanism, orients and spins the third stage in preparation for the final velocity increment. A typical ascent trajectory is shown schematically in Fig. 3. The orientation, produced by the control jets (using residual helium) in the second stage, is to a pitch attitude such that the vehicle is parallel to the earth's surface at 55 degrees South (or North) latitude. After orientation, and before the third stage fires, the third stage and the payload are spun around their roll axis to the angular velocity required to stabilize the vehicle and provide the proper scan rate for the camera.

The firing of the third stage separates it from the second stage and imparts the final velocity increment required to maintain a circular orbit. At third-stage burnout, a small separation device is fired, separating the payload and leaving it, properly oriented and spin-stabilized, in free space. (Appendix E)

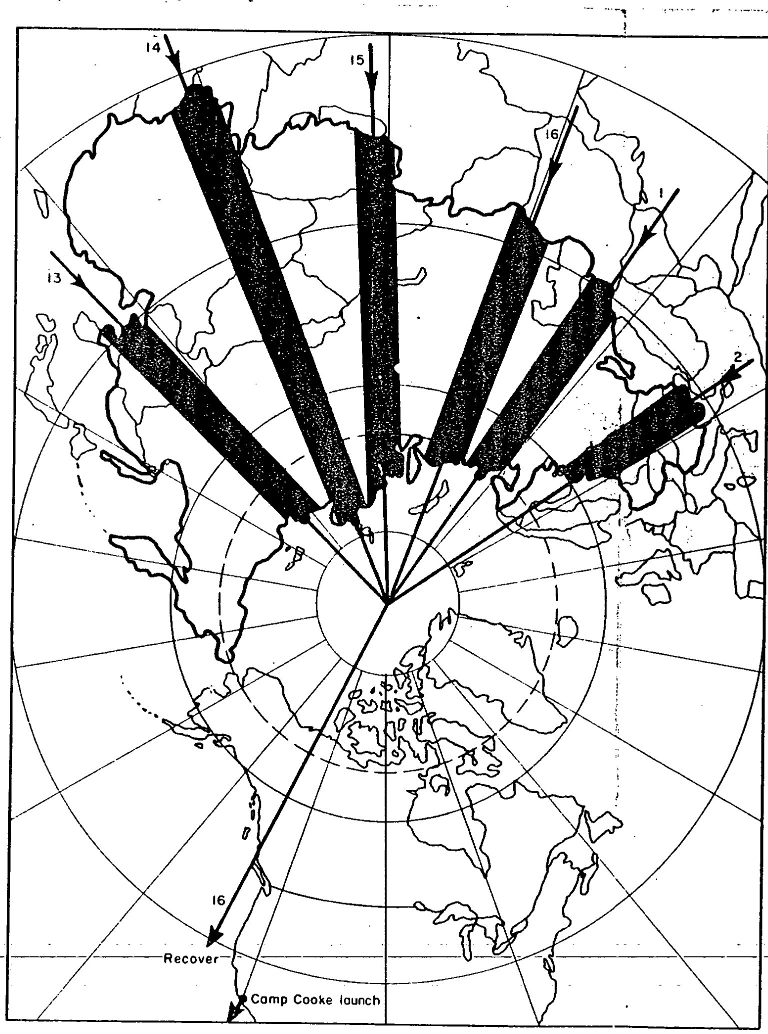


Fig. 4— Ground coverage schematic

V. THE VEHICLE

PAYLOAD STAGE

The weight of the camera and film installation is about 80 pounds. A total payload weight of 300 pounds has been selected, leaving about 220 pounds for payload structural components, re-entry coating, braking-rocket propellants, batteries to operate the film mechanism, beacon and transponder, and for associated gear necessary to operate the camera and recover the package. It might be noted at this point that the power requirement for the camera is conservatively estimated at 100 watt-hours, which can be provided by a few pounds of batteries. Table 1 summarizes the payload-stage weights.

Table 1

Payload-Stage Weight Summary (lb)

Group A -	Photographic Installation		80
	Camera	38	
	Film	10	
	Environment	7	
	Attitude Sensor	10	
	Miscellaneous	15	
Group B -	Structure		110
	Shell	30	
	Fiberglas	80	
Group C -	Recovery System		110
	Impulse Rocket	85	
	Tracking Beacon	16	
	Recovery Beacon	9	
Total Payload-Stage Weight			300

The payload-stage configuration chosen for this case is a double conical shape with a maximum diameter of 35 inches (see Fig. 2). A symmetrical body was chosen because of the desirability of minimizing possible aerodynamic lift forces at the relatively low orbital altitude. Skin material is assumed to be 0.050 magnesium alloy coated with a layer of fiberglas-plastic combination on the forward end for heat protection. The payload stage includes a braking rocket using about 70 lb of propellants.

BOOSTER CONFIGURATION

The booster combination proposed for the early reconnaissance satellite is the Thor IRBM and the second stage of Vanguard, with a small solid rocket, similar in principle to the third stage of Vanguard, to provide a final orbital increment. The propulsion is thus provided by a liquid-liquid-solid combination. Table 2 is a summary of vehicle weights.

Table 2

Vehicle Weight Summary (lb)

Orbital payload	300
Third stage (solid)	350
Second stage (Vanguard 2nd)	4,730
Initial stage (Thor)	110,892

From a preliminary structural investigation it appears that the Thor airframe and its major components need not be modified for the satellite mission. The Thor autopilot and control system can be used for first-stage guidance, although there is some possibility that this system would require some modification to offset the heavier load on the nose of the Thor and the increased loads during the ascent trajectory. For the satellite mission, the Thor inertial guidance system will not be required.

The basic airframe of the Vanguard second stage need not be modified for this application, with the exception of the aft interconnect structure, which will have to be designed to mate the 32-in. diameter Vanguard with the 64-in. diameter Thor nose cone. The guidance system of the second-stage Vanguard would be used in conjunction with G.E. (107A) components.

The size of the third stage has not been optimized. However, the Vanguard third stage is characteristic, in concept at least, of the type of solid propellant rocket that would be required for the present application. (Appendix F)

VI. TRACKING AND RECOVERY

Tracking will be required for essentially three reasons: to determine the orbit accurately enough for coordination of photographic data; to trigger the braking rocket at the proper time for the descent; and to establish the descent path so that the impact point can be located.

The number of trackers required and the spacing between them is dictated partly by the guidance accuracy. To insure against guidance inaccuracies in launching, it is proposed that two or three trackers be used in an arrangement which places them generally with about a 200-mi separation on a line normal to the orbit.

A second factor that must be considered in determining the number and spacing of trackers is the deterioration of tracking accuracy at low angles above the horizon. It is highly important that the satellite pass close enough to at least one station so that sufficient tracking data can be obtained at angles of elevation greater than about 20 degrees. For a nominal satellite altitude of 142 mi, this requires that the satellite pass within roughly 5 degrees of the station, or within a ground range of about 350 mi. Again, two or three trackers at intervals of 200 mi normal to the orbit are dictated.

Because the satellite is to be placed on a polar orbit, these objectives can be met by a small number of trackers near one of the poles. It seems advisable to locate the tracking stations, say three of them, at a high northern latitude such as in Alaska or Canada. Spacing should be about 200 mi in longitude.

Tracking data would be in the form of two angles and a range to permit orbit prediction. The use of range information considerably relaxes the requirement for angular accuracy. To obtain range, a transponder in the satellite is required. (Appendix G)

Descent from orbit is achieved by the command firing of a braking rocket in the satellite. Assume that the satellite is coming over the pole, that it is picked up by trackers in the north, and that an impact point in the Pacific is desired. The braking rocket is then fired forward and upward, imparting a downward and backward velocity impulse superimposed on the orbital velocity. The resulting velocity vector points downward, so that the vehicle is effectively in a ballistic trajectory comparable to the 'low-angle',i.e., lower-than-optimum, path of a long-range ballistic missile.

Tracking of the vehicle immediately after the beginning of descent establishes a predicted vacuum path. This, together with predicted atmospheric effects, makes it possible to predict the approximate impact area. The vehicle is protected against re-entry heating by a coating of suitable vaporizing material: 80 lb of fiberglass-reinforced plastic, such as is used on advanced designs of the ICBM nose cone and on the Jupiter nose cone, is suggested. Impact survival of the casing, film load, batteries, and beacon is made feasible by the proper selection and arrangement of structural components. Search aircraft are used to find and recover the payload. This means that the radio beacon must operate after water impact, and possibly that some type of dye marker should be released upon impact. (Appendix H)

VII. INTELLIGENCE PAYOFFS

Photographs produced by the system just described should enable us to do a useful reconnaissance job at Level A, over areas measuring millions of square miles. The scales and resolution that will be possible are comparable to those obtained with certain kinds of photographic charting equipment standard on Air Force reconnaissance aircraft today. They will make it possible to identify major railroads, highways, and canals. Urban centers, industrial areas, airfields, naval facilities, seaport areas, and the like can be seen. Very likely, defense missile sites of the sort found around the Moscow area will also be identifiable. Thus, with repeated surveillance, it will be possible to find new major installations, perhaps to learn something about patterns of use of Soviet ICBM systems, and certainly to obtain clues for the direction of other, higher-resolution systems that can provide more detailed, accurate identification. (Appendixes B and J)

VIII. GROWTH POTENTIAL

Clearly, the major emphasis of this memorandum is on the easiest and earliest recoverable photographic satellite system. This version, the 12-in., f/3.5 camera using 500 ft of 5-in. film, and based on the Thor booster, is seen as the first of a series of such systems. This payload is capable of photography at Reconnaissance Level A in adequate detail. As the system is proved out, as confidence is gained in satellite operation, and as environmental constraints and intelligence problems become better understood, longer focal-length lenses can be introduced. The first system would be followed by a 36-in. focal-length camera using 1500 ft of 9-in. film. Currently it appears that this payload can also be put on orbit using a Thor-type booster, with a maximum payload weight of about 300 lb.

This second system should provide reconnaissance at Level B, giving adequate detail over areas of hundreds of thousands of square miles. Eventually this system could evolve into one using a 10-ft focal-length lens and about 2500 ft of 18-in. film, based no longer on the Thor but on the Atlas booster, and doing a reconnaissance job at Level C, or over specific point objectives. The time phasing of the several projects should be about as follows: availability of the 12-in. system, one year from date of contract; availability of the 36-in. system in 18 months; and availability of the 10-ft system 36 months after the start of the program (Fig. 5). (Appendix K)

To conclude, it is believed that the type of system proposed here will work, can be available quickly, and will fill a vital military reconnaissance need both in the near and in the distant future.

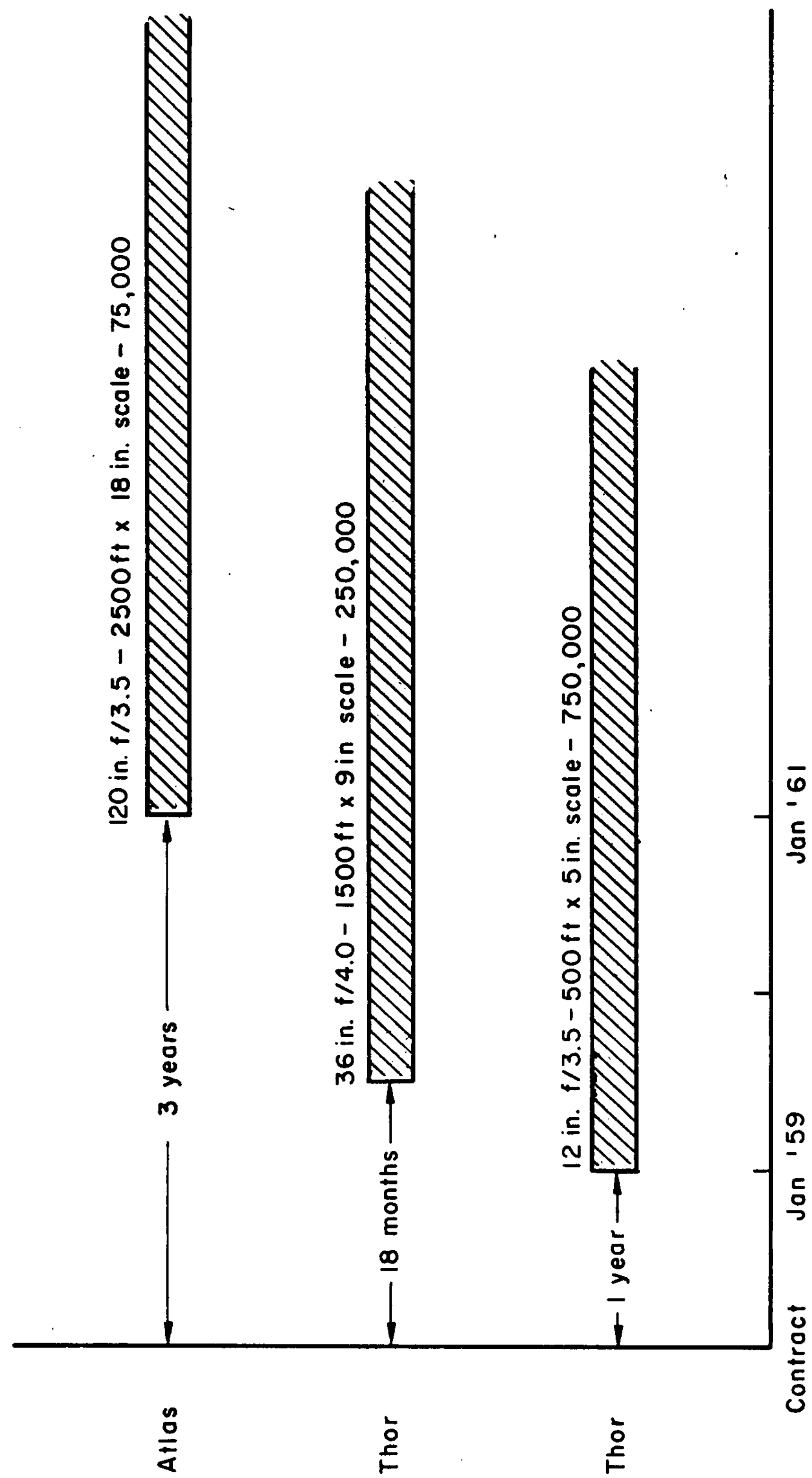


Fig. 5—Growth potential

Appendix A

INTELLIGENCE NEEDS AND MEANS

M. E. Davies and A. H. Katz

Assume that in the forthcoming era we will have a continuing need for both national and military intelligence. It will suffice for present purposes to note that national intelligence needs, though using in large measure the same kinds of information inputs required to satisfy military intelligence needs, are by and large tied to the formation of objectives and policies, and to the supporting actions necessary to implement these objectives.

Military intelligence, on the other hand, finds its needs in the requirements to plan defensive and offensive responses to enemy actions and to ensure adequate development programs to meet estimated threats.

Military intelligence needs can be described under three major categories:

1. Warning
2. Estimate of Capabilities
3. Targeting Information

Warning is a problem of overriding priority. The requirement here is to supply warning of imminence of hostilities. The second category of intelligence requirements will, if met, answer the questions: what does the enemy have, how does he use it, how good is he at using it, where is it, and how many does he have? Targeting information is that information necessary for us to plan our military response to attack. The reconnaissance system described in this report is intended to provide intelligence for categories (2) and (3).

Intelligence data can be supplied by many methods. Certainly a major method is physical-objective reconnaissance using sensors such as photography, radar, infrared, and the like, and airborne photographic reconnaissance emerges as perhaps the principal reconnaissance tool for collecting data about the Soviet Union. It is believed that only high-resolution photographic reconnaissance can meet the requirements for detailed reconnaissance that will exist for the next few years at least.

In attempting to pick the kinds of systems which are preferred for the job of pre-hostilities aerial reconnaissance of the Soviet Union, it is necessary to have an operational concept. It is convenient to define four levels of photographic reconnaissance: A, B, C, and D.

Reconnaissance Level A provides pioneer large-area search. This calls for a system to be operated over areas measured in millions of square miles, and at a photographic scale of roughly 250,000 K (where K is between 1/2 and 2) on Aero Super XX at approximately 10 lines/mm.

High altitude operations which have as their end the securing of detail about small ground objects require resolutions which can perhaps be measured in terms of feet, or dozens of feet, on the ground. It is entirely useless to talk about scale without specifying the level of definition or resolution, and the material on which it is obtained. For this discussion of photographic performance we have chosen to make the numbers consistent with nominal service-obtained resolution (10-15 lines/mm) on standard Aero XX film. While this resolution is often exceeded in practice, it is convenient to use as a reference against which photographic performance is measured. Thus we should be able to take this scale number (250,000 K) for pioneer large-area search against millions of square miles, and investigate the usefulness of a proposal at a different scale and at a different level of resolution. Very roughly, resolution and scale are interchangeable over small excursions. Thus a scale of 1,000,000 at 40 lines/mm should be approximately as good as a scale of 250,000 at 10 lines/mm--but if a choice is available, the latter is certainly to be preferred.

Level B is entitled 'limited area search,' proceeding at a scale number five times smaller (scale therefore five times larger) of 50,000 K, useful against areas measured in hundreds of thousands of square miles. At such a level of reconnaissance the character of many major installations can be detected and identified, aircraft can be seen on airfields, minor lines of communication can be found and plotted, and in general, those items found at Level A can be seen more satisfactorily.

Level C, 'specific point objective photography,' can be accomplished at a scale level of 10,000 K, and is useful against target areas measured in terms of hundreds of square miles. At this level detailed analyses of sites, airfields, industries, and activities can be made.

Level D, 'technical intelligence objective,' at a scale level of 2,000 K, is useful against areas measured in tens of square miles or less. It will serve many if not most of the needs of technical intelligence which can be met by photography at all.

Air Force photography, in some instances, is at present carried out at precisely the scales of Level A. An immediate reference is available in the scale out near the middle of the photographs obtained by the oblique trimet charting installations standard on most Air Force reconnaissance aircraft. It is clear from studying this kind of photography and the resolution obtained, that under these conditions one should be able to see and identify most lines of communication, railroads--certainly major railroads--, highways, canals, urban centers, industrial areas, air fields, naval facilities, seaport areas, and the like. Very likely, the defense missile sites of the sort found around the Moscow area are identifiable under these conditions.

A concept of operations is envisioned whereby all of the Soviet Union is covered at this pioneer level of quality at intervals of say, 6 months to 1 year. With such an operation, new major installations can be detected, patterns of use perhaps found, and certainly hints and clues for the direction of other finer higher-resolution reconnaissance systems can be obtained. The overall reconnaissance capability of the United States must be based upon systems able to operate at each of these levels. The system we are concerned with here is designed specifically to do job A, which occurs first in order of interest and must be done in order to serve as a useful guide for further reconnaissance jobs.

It is extremely difficult, if not impossible, to take a given number, such as ground resolution of--in the case of this proposal, 60 feet--and say specifically what can be seen. The conditions of observation are so variable, as are the illumination, the contrast, the context, and many other important factors that determine detection and identification, so that to specify ground resolution alone is insufficient. At least one photograph is available, taken with a six-inch lens from an altitude of about 150 miles, with resolution estimated to be about 10 lines/mm in the focal plane, therefore giving a ground resolution of about 500 feet on the ground. On this particular photograph, major railroads show up clearly with even casual observation; two military airfields are easily seen with runways clear and distinguished; and major streets in a nearby city are fairly easily resolvable and can be plotted. (This exemplifies the phenomenon of long lines being more easily detected than small square objects.)

The number of ways of conducting overflight operations over the Soviet Union for the next few years are sharply limited in number. Satellites and aircraft almost exhaust the list of possibilities, and for present purposes will be sufficient. For numerous reasons, including political palatability, vulnerability, and the like, it is unrealistic to conceive of wholesale cyclic mapping of the Soviet Union by aircraft. Aircraft are superb for performing reconnaissance at Level C and D, and, properly used, should be confined to that purpose, furnishing a complementary system to whatever system or systems operate at Levels A and B.

It is realistic to conceive of finding large military installations, or other installations in the process of being built, using systems performing at Level A or B. At such levels, it is certainly possible to get clues of sufficient interest to warrant dispatching a system operating at Levels C and D to verify, confirm, or further inspect these operations. This is particularly true for the urgent problem of finding, identifying, and plotting on a map, Soviet Union missile sites.

In conclusion, reconnaissance at Level A is seen as a fundamental and 'first-thing-first' job. It will supply a matrix in which other data can be imbedded, and it will furnish a planning guide for future reconnaissance. In and of itself it will sort out profitably searched areas from worthless areas, and it will locate and chart major military and industrial operations as well as help determine patterns of use.

Appendix B

CAMERA DESIGN

M. E. Davies and A. H. Katz

The essence of the present proposal is to put a camera-carrying satellite on orbit approximately 150 miles above the earth, operate this camera system over the Soviet Union for approximately one day, and bring the film back to the United States for military analysis. This appendix discusses the design and operation of the camera and some of the problems of the satellite mission.

Panoramic cameras for both ground and aerial use are certainly not new. This particular camera design was inspired by a series of proprietary proposals by the Fairchild Camera and Instrument Corporation. Extensive conversations about these systems were held with F.P. Willcox, Vice President of Fairchild, starting in the spring of 1957. Their Engineering Proposal No. 431 describes a 12-in. panoramic for the Ryan tip-pod.* As suggested by RAND for use in the satellite, panoramic scanning will be accomplished by rolling the entire satellite stage rather than rotating the camera. For the satellite application, the lens will be mounted behind a quartz window in the carrier, and will image the ground on a fixed slit in front of the focal plane. Since this fixed slit serves as a shutter, the only moving parts will be the film transport. During the part of the rotational period in which the ground is not being photographed, a measured length of film

*After initially considering the use, in a satellite, of axially oriented cameras employing 45° mirrors, as in the Fairchild idea, it occurred to the authors that the mirror could be removed and the camera tipped to the vertical position. This makes it similar in principle to a camera used for transverse panoramic photography by Col. R. W. Philbrick at Boston University in about 1948. This latter camera was modified from a standard USAF S-7 camera; in-flight photographs made with this 'Whirling Dervish' were exhibited to the U.N. in 1955, as part of an 'Open Skies' exhibit.

will be rolled off the supply spool in readiness for the next exposure. At the same time, the last exposure will be wound on the take-up spool. If desired, image motion compensation can be accomplished by canting the slit, so that the film motion vector corresponds to the average image motion vector.

The camera for the large-area search mission will have a 12-in. f/3.5 lens and carry 500 ft of 5-in. film. At a design altitude of 750,000 ft (roughly 150 miles), a sweep angle of 93° corresponds to a strip width of about 300 mi. The format size is 20 in. x 4-1/2 in. with a scale at the center of 750,000, and an average scale of about 1,300,000 at the edge. The ground coverage at the center is 53.3 mi and 77.5 mi at the edge. Each photograph will cover some 18,000 sq mi.

The missile roll-rate is 18.2 rpm with the camera making an exposure every third revolution. This gives 9.9 sec between start of successive exposures, 10 per cent forward ground overlap in the vertical on successive frames, and a required film speed of 22.9 in. per second. The design shutter speed of 1/4000 sec dictates a slit width of .0057 in. which, from the standpoint of construction, is quite feasible. This slit would be canted to correct for .41 in. per sec forward image motion at the center of the frame.

The 500 ft of film will permit 300 exposures; this corresponds to a ground path length of 14,300 mi and a total coverage of 4,000,000 sq mi. It is estimated that this camera will weigh about 50 lb; the peak power load will be about 100 watts. The film and spool will weigh about 10-1/2 lb.

It is likely, and desirable, that a special 12-inch lens should be designed for this camera. This will not be too difficult, for the laboratory performance required on the film will be about 60 lines/mm.

There exist today several lens designs which can be scaled up and several which could be scaled down to do this job. The essentially narrow angle of coverage, the moderate speed, and the fairly short (by aerial reconnaissance standards) focal length are factors ensuring the emergence of an acceptable lens in a reasonable time.

The exposure time, 1/4000 sec, is based on the choice of a rather new film emulsion used for aerial photographic purposes—Eastman Kodak S.O. 1166 (this was the number under which this film was known up until very recently when it was given the name, Plus-X Aerecon). The choice of lens speed (f/3.5) and shutter speed are entirely consistent with this film choice.

We can examine the image motion that may occur during the time a photograph is taken with this panoramic system rather easily. First, of course, there is forward image motion; in this case, the motion caused by imaging a ground velocity of about 25,000 ft/sec at a scale of about 750,000. A speed of 25,000 ft/sec for 1/4000 sec results in a ground motion of 6 feet during the exposure, which, when reduced by the appropriate scale factor, gives the amount of image displacement. As noted above, the film—Plus-X Aerecon—was chosen such that it would have enough speed to permit exposure at 1/4000 sec at f/3.5. The logical question at this point is whether such a film is capable of imaging aerial photographs at sufficient resolution to permit examination of ground detail at a level consistent with military requirements. The answer is yes. The major differences between Plus-X Aerecon and standard Aero Super XX are that Plus-X Aerecon has more resolution and much better graininess characteristics, thus permitting the examination of finer detail with more confidence than Aero Super XX.

This system is not intended to obtain microscopic resolution at, say, levels of 100 lines/mm. Rather, the goal aimed at, and the goal believed to be fairly easily attainable, is a modest 40 lines/mm from the air. This compares with advanced reconnaissance systems now in being. A statistic commonly used in describing aerial reconnaissance systems is ground resolution. Ground resolution is simply the ground dimension which corresponds to one line of resolution in a focal plane. In this case, with a 12-inch focal length lens and 40 lines/mm, clearly the width of a ground element produced or projected back on the ground through the lens system is 1/40 mm as seen from 12 inches (300 mm). This is 1/12,000 of the altitude, or approximately 60 feet. Thus the element of resolution with which we are concerned is about 60 feet.

It is for this reason that residual motions during the exposure, which are a small fraction of the 60-feet resolution, can, by and large, be ignored, especially when these fractions are of the order of 10 per cent or less. Hence, the 6 feet or so of forward motion produced during the exposure time of 1/4000 sec while the vehicle is moving at 25,000 ft/sec, is 1/10 the basic resolution element and can, in this case, be easily ignored. On the other hand, it is perfectly possible to put in an average compensation for forward ground speed by properly orientating the slit mechanism.

By the same token the tolerance or sensitivity of this system to angular motion and angular rates can be investigated. The amount of motion or blurring produced on the film during an exposure is simply the uncompensated motion, i.e., uncompensated image speed times the exposure time. Assume a residual speed of, say, 1 inch per second and an exposure time of 1/4000 second; the residual image motion is therefore 1/4000 inch, which is approximately 1/160 mm, considerably less than that tolerable by a 40 lines/mm standard.

In general, uncompensated image motions other than that caused by forward motion of the vehicle arise from angular rates of one sort or another. An angular rate of 5 degrees/sec gives rise to an uncompensated image speed in the focal plane of a 12-inch lens, of 1 inch/sec. Considering the exposure time of 1/4000 sec, this gives a residual motion of about 1/4000 in., which again is about 1/160 mm, and is negligible.

This example is illustrative only. Angular rates of this magnitude are not anticipated for this system. The other major, or potentially major, source of blurring occurs if the film is pulled through at a rate different from that demanded by the spin rate of the vehicle. Table 1 shows the relationship between spin rate, focal length, length of film, and other parameters of this 12-inch system, as well as other systems which we envision as follow-on developments. It is intended to generate a film speed in the transverse direction of about 24-in./sec. This is based on a spin rate of about 18.2 rpm.

Table I

CAMERA SUMMARY

Focal Length (in.)	12"	36"	120"
Strip Width (s. miles)	300	300	100
Sweep Angle	93° 8'	93° 8'	38° 47'
Format Size	4-1/2" x 19-1/2"	9" x 58-1/2"	18" x 81-1/4"
Length of Film Carried (ft)	500	1500	2500
Number of Frames	300	305	365
Frequency of Exposures (sec)	9.9	6.6	4.0
Spin Rate (rpm)	18.2	18.2	15.2
Revolutions Between Exposures	2	1	0
Film Speed (in./sec)	22.9	68.6	190.6
Forward Motion (in./sec)	.4	1.2	4.1

Suppose the film rate is off by 10 per cent, either too high or too low from that demanded by these constraints. In this case, the differential image speed—uncompensated image speed—is again approximately 10 per cent of the 24-in./sec or 2-in./sec. This uncompensated image speed would give at most 1/80 mm of uncompensated motion in the single direction. This is not enough to affect seriously the overall resolution. (It is not anticipated that the film rate will be as much as 10 per cent off in this system.)

It may be of interest to examine rough power requirements for this type of camera. The actual camera operation will occur during a 90-degree, essentially vertical portion of a revolution, 1/4 revolution. The actual photography then, will be accomplished during this 1/4 revolution out of every three revolutions. Thus the camera is drawing peak load for 1/12 of the time of the period of photography. It is estimated that peak power requirements for this operation, based on performance requirements of somewhat similar cameras, will be no more than 100 watts. Between pictures, that is for the other 11/12 of the cycling time, the camera will be metering film out in readiness for the next exposure, and winding up on the take-up spool the film taken on the previous exposure. This operation consumes much less power than does the main job of pulling film through at 24-in./sec.

The number of actual flight-line miles contemplated in this operation is about 14,000. This corresponds to about seven 2,000-mile passes over the Soviet Union, or six passes of somewhat greater length. The total flight-line miles of the vehicle is approximately the circumference of the earth times the number of passes around. This is about 400,000 miles.

Thus percentage of on-time of the total camera system is approximately 14/400 of the total life of the vehicle, here taken as one day. This is 3-1/2 per cent of the one day, or 50 minutes.

Neglecting the fact that the camera is drawing peak power for only 1/12 of this total on-time, assume that it is drawing this 100 watts on a continuous basis for the 50 minutes. The total amount of power required for the camera is less than 100-watt hours, an amount of energy that can be furnished by a few pounds of batteries.

Appendix C

ATTITUDE STABILIZATION

T. B. Garber

By spinning a vehicle about its longitudinal axis, the attitude of the body may be stabilized with respect to an inertial reference. It is of interest to examine the effect upon the angular motion of the body of external, applied torques. Such torques may arise from a number of sources: aerodynamic forces, the motion of components within the vehicle, or disturbances during propulsion. Figures 1 and 2 define some of the variables of interest.

In order to simplify the analysis, a circular orbit has been assumed. The rotational equations of motion of the vehicle, in body axes are:

$$\dot{\omega}_x + \frac{I_z - I_y}{I_x}\,\omega_y\,\omega_z = \frac{M_x}{I_x} \qquad (a)$$

$$\dot{\omega}_y + \frac{I_x - I_z}{I_y}\,\omega_x\,\omega_z = \frac{M_y}{I_y} \qquad (b) \qquad (1)$$

$$\dot{\omega}_z + \frac{I_y - I_x}{I_z}\,\omega_x\,\omega_y = \frac{M_z}{I_z} \qquad (c)$$

From Fig. 2 the relation between angular rates in body axes and the rate of change of the orientation angles may be deduced.

$$\dot{\theta} = \omega_y \cos\psi - \omega_z \sin\psi \qquad (a)$$

$$\dot{\gamma}\cos\theta = \omega_y \sin\psi + \omega_z \cos\psi \qquad (b) \qquad (2)$$

$$\dot{\psi} = \omega_x + \tan\theta\left[\omega_y \sin\psi + \omega_z \cos\psi\right] \qquad (c)$$

The first disturbances that will be considered are those due to residual control system errors. With a constant spin rate, ω_x, about the

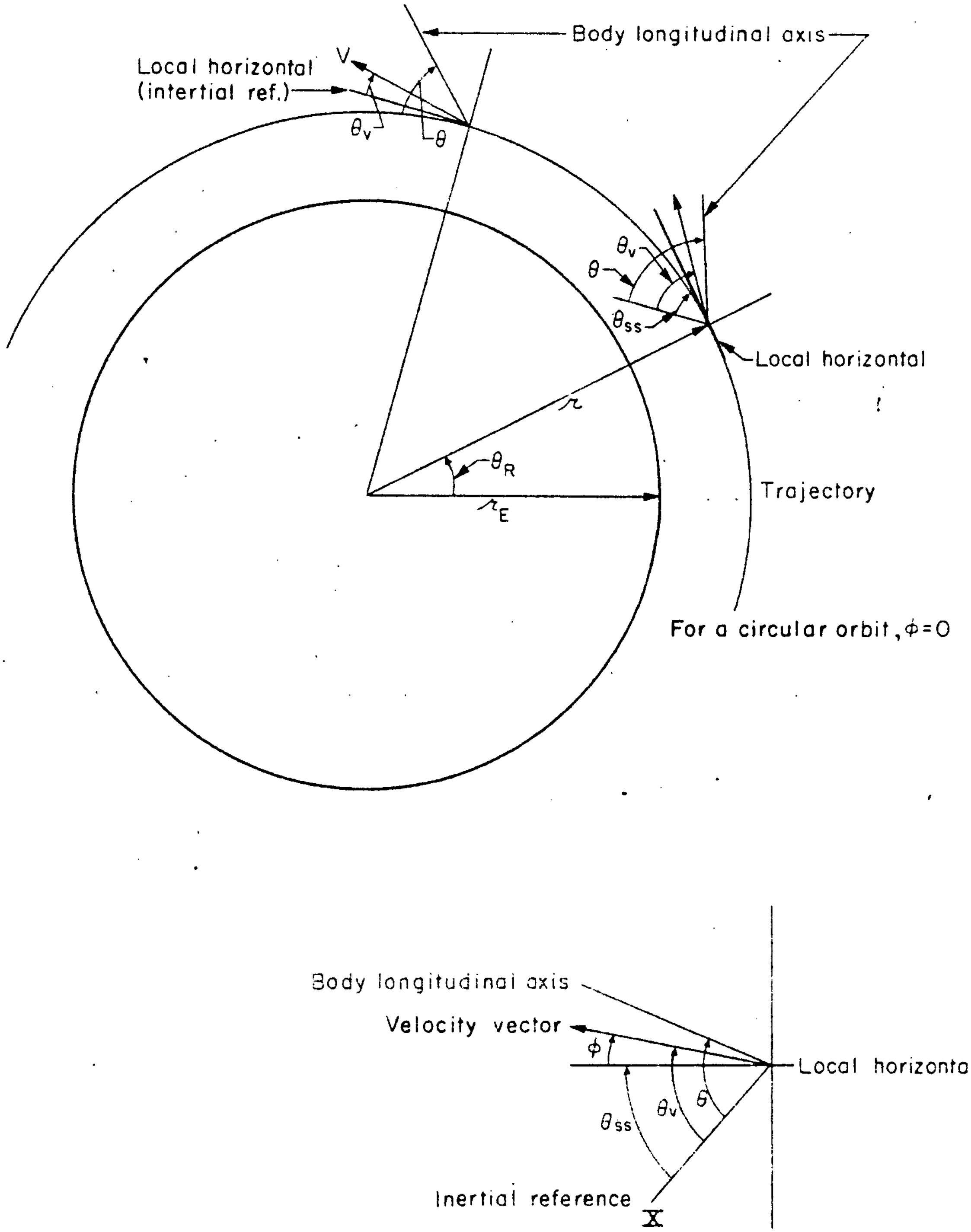
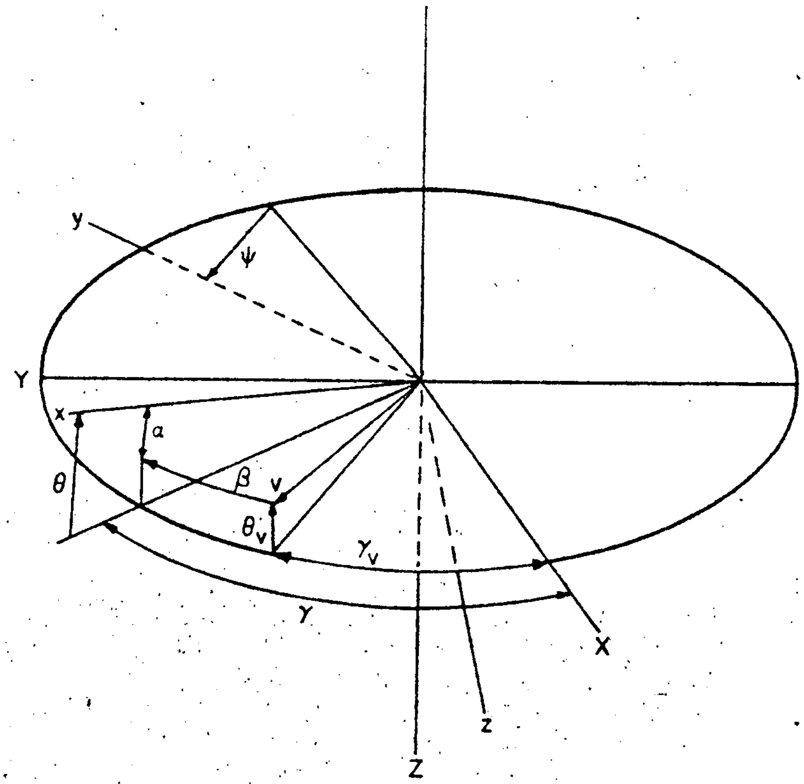


Fig. C-1 — Planar trajectory schematic



X, Y, Z, Inertial system
x Longitudinal axis of the vehicle

Fig. C-2—Orientation angles indicating angular positions of the velocity vector and longitudinal axes with respect to an inertial reference longitudinal axis, and with no disturbing torques, Eq. (1, b and c) have the following solutions:

$$\omega_y = \omega_{y_o} \cos R_z \omega_x t + \omega_{z_o} \sin R_z \omega_x t \tag{3}$$

$$\omega_z = \omega_{z_o} \cos R_z \omega_x t - \omega_{y_o} \sin R_z \omega_x t \tag{4}$$

where $I = I_z = I_y$ and $R_z = \frac{I - I_x}{I}$

Introducing Eqs. (3) and (4) into Eq. (2) yields

$$\theta = \theta_o + \epsilon_\theta - \frac{\epsilon_{\dot{\gamma}}}{R_x} + \frac{\epsilon_{\dot{\gamma}}}{R_x} \cos R_x t + \frac{\epsilon_{\dot{\theta}}}{R_x} \sin R_x t \tag{5}$$

$$\gamma = \epsilon_\gamma + \frac{\epsilon_{\dot{\theta}}}{R_x} - \frac{\epsilon_{\dot{\theta}}}{R_x} \cos R_x t + \frac{\epsilon_{\dot{\gamma}}}{R_x} \sin R_x t \tag{6}$$

where $R_x = \frac{I_x \omega_x}{I}$, and the residual initial errors are denoted by ϵ with the appropriate subscript.

In obtaining Eqs. (5) and (6), it has been assumed that θ and γ are small angles, and that their product and the products of their time derivatives may be neglected. The validity of this assumption depends upon the magnitude of ω_x in Eq. (2,c) as compared to the remaining terms on the right hand side of the equation.

The angle of attack, α, is of interest since it indicates the departure of the longitudinal axis of the vehicle from the local horizontal. With the assumption that, over the range of θ_R of interest, γ and γ_v are small angles, the rigid body angular motion with respect to the velocity vector in the absence of aerodynamic or other disturbing torques may be readily determined. (See Figs. 1 and 2.)

$$\alpha = + \dot{\theta}_{ss}t + \frac{\epsilon_{\dot{\beta}_o}}{R_x}\cos R_x t + \frac{\epsilon_{\dot{\alpha}}}{R_x}\sin R_x t + (\alpha_o + \epsilon_\alpha - \frac{\epsilon_{\dot{\beta}}}{R_x}) \qquad (7)$$

$$\beta = \frac{-\epsilon_{\dot{\alpha}}}{R_x}\cos R_x t + \frac{\epsilon_{\dot{\beta}}}{R_x}\sin R_x t + \epsilon_\beta + \frac{\epsilon_{\dot{\alpha}}}{R_x} \qquad (8)$$

where α_o is the desired attitude of the body at the initiation of the problem, and ϵ_α, ϵ_β, $\epsilon_{\dot{\alpha}}$, and $\epsilon_{\dot{\beta}}$ are the residual errors in α and β and the rates of change of those angles at the end of the orientation control period.

As would be expected, with all of the control errors equal to zero, the body longitudinal axis is stationary with respect to an inertial reference. However, an examination of the above equations indicates that with rate errors present, the usual precessional motion of a spinning body will occur.

Consider now the resultant error in θ and γ caused by this precession. For the error in $\dot{\theta}$, $\epsilon_{\dot{\theta}}$ values of 1°/sec to 3°/sec will be assumed. This performance is comparable to that obtained in the attitude control of current vehicles.* A similar value will be assumed for $\epsilon_{\dot{\gamma}}$. The angular position errors, ϵ_θ and ϵ_γ, might be as large as 2°. With θ and γ small angles, the vehicle orientation error due to residual rate errors is:

*Buchheim, R. W., 'Lunar Instrument Carrier--Attitude Stabilization,' RM-1730, June 4, 1956 (Unclassified).

$$\Delta A_{\theta,\gamma} = 1.41 \left\{ \left[\left(\frac{\epsilon_{\dot{\theta}}}{R_x} \right)^2 + \left(\frac{\epsilon_{\dot{\gamma}}}{R_x} \right)^2 \right] (1 - \cos R_x t) \right\}^{1/2} \tag{9}$$

Figure 3 indicates the manner in which the envelope of $\Delta A_{\theta,\gamma}$ varies as a function of the parameter R_x.

At the end of the control and stabilization period the vehicle is boosted to orbital speed. With the thrust axis misaligned by a small angle, δ, a disturbing torque is applied to the craft.

$$M_y = T \cdot \ell_x \cdot \delta \tag{10}$$

where T is the thrust, and ℓ_x is the moment arm.

With zero initial conditions, Eq. (1, b and c) now have the following solutions:

$$\omega_y = \frac{T \ell_x \delta}{R_z \omega_x I} \sin R_z \omega_x t \tag{11}$$

$$\omega_z = \frac{T \ell_x \delta}{R_z \omega_x I} (\cos R_z \omega_x t - 1) \tag{12}$$

where $0 \leq t \leq t_B$, and $R_z = \frac{I - I_x}{I}$.

It is assumed that the thrust is applied as a step function. Thus, Eqs. (11) and (12) are valid up to the end of the burning period, t_B. When $t > t_B$, ω_y and ω_z have the following form:

$$\omega_y = \frac{T \ell_x \delta}{I R_z \omega_x} \left[\sin R_z \omega_x t_B \cos R_z \omega_x t + (\cos R_z \omega_x t_B - 1) \sin R_z \omega_x t \right] \tag{13}$$

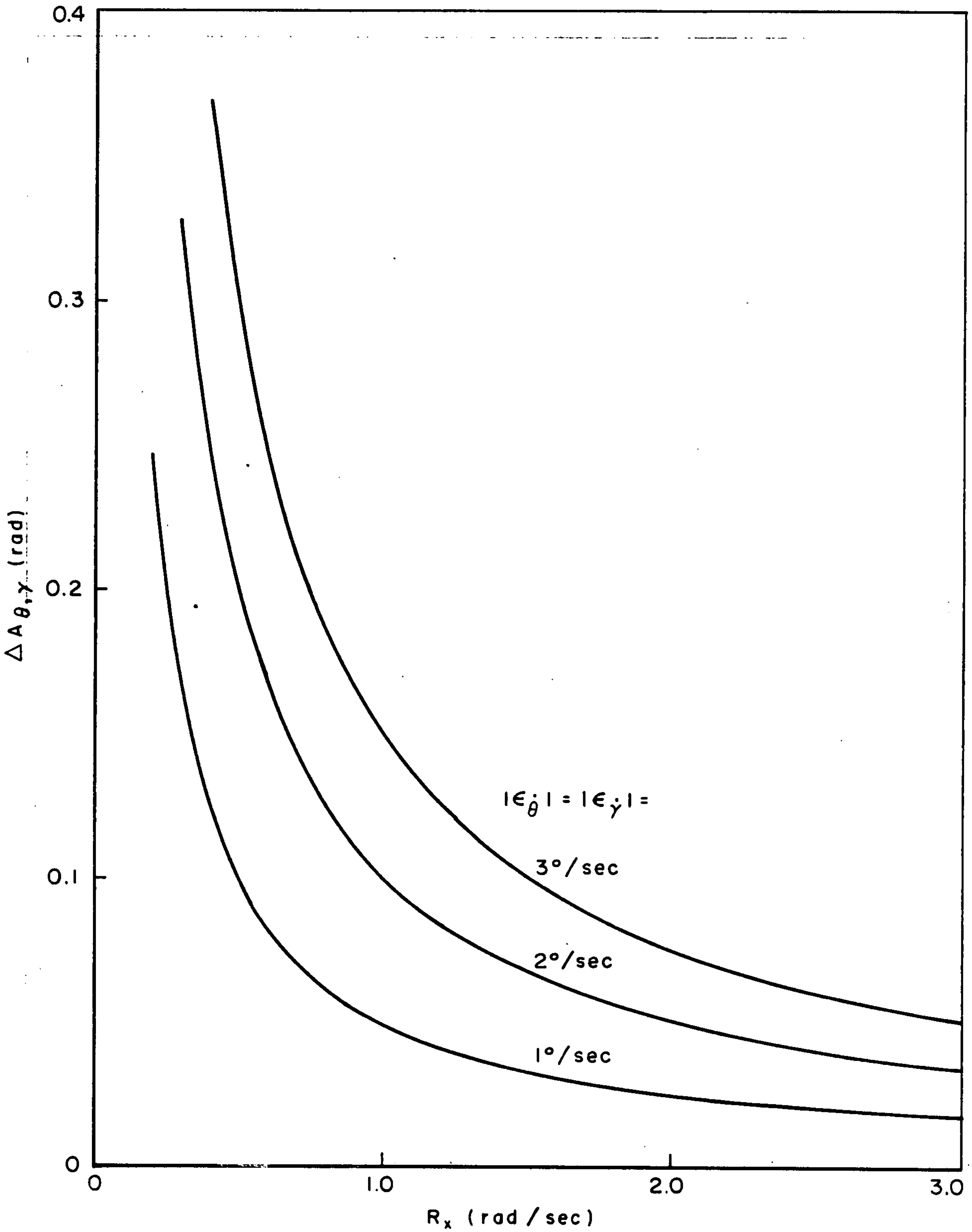


Fig. C-3 — Total attitude error as a function of spin rate

$$\omega_z = \frac{T\,\ell_x\,\delta}{I\,R_z\omega_x}\left[(\cos R_z\omega_x t_B - 1)\cos R_z\omega_x t - \sin R_z\omega_x t_B \sin R_z\omega_x t\right] \quad (14)$$

where $t > 0$. (Zero time redefined at the end of burning.)

Introducing Eqs. (11) and (12) into Eq. (2, a,b and c) yields

$$\dot{\theta} = \frac{T\,\ell_x\,\delta}{I\,R_z\omega_x}\,p_1 \qquad (a)$$

$$\dot{\gamma}\cos\theta = \frac{T\,\ell_x\,\delta}{I\,R_z\omega_x}\,p_2 \qquad (b) \quad (15)$$

$$\dot{\psi} = \omega_x + \tan\theta\,\frac{T\,\ell_x\,\delta}{I\,R_z\omega_x}\,p_2 \qquad (c)$$

where $0 \leq t \leq t_B$ and where

$$p_1 = \sin R_z\omega_x t \cos\psi + (1 - \cos R_z\omega_x t)\sin\psi$$

$$p_2 = \sin R_z\omega_x t \sin\psi - (1 - \cos R_z\omega_x t)\cos\psi$$

In the case of Eq. (15), the validity of a small angle solution depends upon the magnitude of $T\,\ell_x\,\delta/I\,R_z\omega_x$. It should be noted that this term may be reduced by reducing the thrust and increasing the burning time such that the total impulse remains constant.

A procedure that may be followed in obtaining solutions to Eq. (15, a, b and c) is to expand ψ in terms of a Taylor series in one of the parameters of $T\,\ell_x\,\delta/I\,R_z\omega_x$. Thus

$$\psi(t,\delta) = \psi_{\delta=0} + \left(\frac{\partial\psi}{\partial\delta}\right)_{\delta=0}\delta + \frac{1}{2}\left(\frac{\partial^2\psi}{\partial\delta^2}\right)_{\delta=0}\delta^2 + \ldots \qquad (16)$$

with a little manipulation, θ may be eliminated from Eq. (15, b and c)

$$\ddot{\psi} = \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x}\right)^2 p_1\, p_2 + (\dot{\psi} - \omega_x)^2\, \frac{p_1}{p_2} + (\dot{\psi} - \omega_x)\, \frac{\dot{p}_2}{p_2} \qquad (17)$$

From Eq. (17) the first three terms of the expansion of Eq. (16) may be determined. Thus:

$$\psi_{\delta=0} = \omega_x t \qquad (a)$$

$$\left(\frac{\partial \psi}{\partial \delta}\right)_{\delta=0} = 0 \qquad (b)$$

$$\left(\frac{\partial^2 \psi}{\partial \delta^2}\right)_{\delta=0} = \left(\frac{T\, \ell_x}{I\, R_z \omega_x}\right)^2 \Bigg\{ \frac{t\,(I + I_x)}{\omega_x I_x} + \frac{2}{\omega_x^2} \Bigg[\frac{I\,(I_x - I)}{I_x^2} \sin R_x t$$

$$+ \left(\frac{I - I_x}{I_x}\right) \sin \omega_x t - \left(\frac{I\,(I_x + I)}{2\, I_x\, (I - I_x)}\right) \sin R_z \omega_x t \qquad (18)$$

$$- \frac{I}{2\, I_x} \sin (\omega_x + R_x)\, t + \frac{I^2}{4\, I_x^2} \sin 2\, R_x t \qquad (c)$$

$$+ \frac{1}{4} \sin 2\, \omega_x t \Bigg] \Bigg\}$$

An examination of Eq. (18, a,b, and c) indicates that the first effect of the misalignment torque appears in the coefficient of δ^2, and consists of a term linear in time plus an oscillating component.

$$\psi \cong \left[\omega_x + \frac{1}{2} \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^2 \frac{(I + I_x)}{\omega_x I_x} \right] t + \frac{1}{\omega_x^{\,2}} \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^2 \psi_{osc.} \qquad (19)$$

Introducing Eq. (19) into Eq. (15,a) and integrating yields

$$\theta = A_{R_x} \left\{ \cos \left[R_x + \mu \right] t - 1 \right\}$$

$$+ A_{\omega_x} \left\{ 1 - \cos \left[\omega_x + \mu \right] t \right\}$$

$$+ \frac{1}{2\omega_x^{\,3}} \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^3 f_1 (\psi_{osc.}) \qquad (20)$$

where

$$A_{R_x} = \frac{\dfrac{T\, \ell_x\, \delta}{I\, R_z\, \omega_x}}{R_x + \dfrac{1}{2} \left(\dfrac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^2 \dfrac{(I + I_x)}{I_x\, \omega_x}}$$

$$A_{\omega_x} = \frac{\dfrac{T\, \ell_x\, \delta}{I\, R_z \omega_x}}{\omega_x + \dfrac{1}{2} \left(\dfrac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^2 \dfrac{(I + I_x)}{I_x\, \omega_x}}$$

$$\mu = \frac{1}{2} \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^2 \frac{(I + I_x)}{I_x \omega_x}$$

From Eq. (20) it can be seen that the disturbing torque influences both the amplitude and the period of the pitch oscillation.

Combining Eq. (15,b and c) yields:

$$\dot{\gamma} = \pm \left\{ \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^2 p_2^{\,2} + (\dot{\psi} - \omega_x)^2 \right\}^{1/2} \tag{21}$$

Equation (21), upon consideration of Eq. (18) and Eq. (19), may be reduced to the following form:

$$\dot{\gamma} \cong \frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} p_2 + \frac{1}{2\omega_x^{\,2}} \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^3 \left[\frac{I}{I_x} (\cos R_x t - 1) + (1 - \cos \omega_x t) \right]^2 p_2 \tag{22}$$

Integrating Eq. (22) yields:

$$\gamma \cong A_{R_x} \sin \left[R_x + \mu \right] t \tag{23}$$

$$- A_{\omega_x} \sin \left[\omega_x + \mu \right] t + \frac{1}{2\omega_x^{\,3}} \left(\frac{T\, \ell_x\, \delta}{I\, R_z \omega_x} \right)^3 f_2 \left(\psi_{osc.} \right)$$

If terms up to δ^2 are retained, then the pitch and yaw errors due to thrust misalignment at the end of the burning period are:

$$\Delta\theta_\delta = A_{R_x} \left[\cos (R_x + \mu)\, t_B - 1 \right] + A_{\omega_x} \left[1 - \cos (\omega_x + \mu)\, t_B \right] \tag{24}$$

and

$$\Delta\gamma_\delta = A_{R_x} \sin (R_x + \mu)\, t_B - A_{\omega_x} \sin (\omega_x + \mu)\, t_B \tag{25}$$

An examination of Eqs. (24) and (25) indicates that, for a <u>known</u> misalignment angle, $\Delta\theta_\delta$ and $\Delta\gamma_\delta$ may be constrained to be zero at t_B for any particular vehicle design, regardless of the value of A_{R_x} and A_{ω_x}. Thus,

$$R_x + \mu = \frac{n\pi}{t_B} \tag{26}$$

$$\omega_x + \mu = \frac{(n+2)\pi}{t_B} \tag{27}$$

where n is an even integer.

Introducing the expression for μ given with Eq. (20) into Eq. (27) yields:

$$\omega_x^4 - \frac{(n+2)\pi}{t_B}\omega_x^3 + \frac{1}{2}\left(\frac{T\,\ell_x\,\delta}{I\,R_z}\right)^2\left(\frac{I + I_x}{I_x}\right) = 0 \tag{28}$$

The ratio, $\frac{I_x}{I}$ may be found in terms of ω_x and t_B by subtracting Eq. (26) from Eq. (27). Thus,

$$\omega_x^2 - \frac{(n+2)\pi}{t_B}\omega_x + \left(\frac{T\,\ell_x\,\delta\,t_B}{2\pi\,I}\right)^2\left[\frac{\omega_x t_B - \pi}{\omega_x t_B - 2\pi}\right] = 0 \tag{29}$$

An approximate solution of Eq. (29) may be found by expanding ω_x in a Taylor series in δ.

$$\omega_x = \frac{(2+n)\pi}{t_B} - \frac{t_B}{(n+2)\pi}\left(\frac{n+1}{n}\right)\left(\frac{\nu\,\ell_x\,\delta}{2\pi\,I}\right)^2 \tag{30}$$

where ν is the total impulse.

The ratio $\frac{I_x}{I}$ is

$$\frac{I_x}{I} = 1 - \frac{2\pi}{\omega_x t_B} = \frac{n\pi - \frac{t_B^2}{(n+2)\pi}\left(\frac{\nu\,\ell_x\,\delta}{2\pi\,I}\right)^2\left(\frac{n+1}{n}\right)}{(2+n)\pi - \frac{t_B^2}{(n+2)\pi}\left(\frac{\nu\,\ell_x\,\delta}{2\pi\,I}\right)^2\left(\frac{n+1}{n}\right)} \tag{31}$$

Figure 4 is a plot of the required ω_x and I_x/I as a function of the burning time for the case in which n is 2.

The thrust misalignment angle δ is, of course, only known in a statistical sense. Figure 5 shows the variation of the errors in θ and γ as a function of δ. In this particular case, the system has been tuned such that the errors in pitch and yaw would be zero at the end of a 5.9 second burning period.

Pitch and yaw errors would be expected at the end of the burning period due to errors in the spin rate, the burning time, and in other parameters, such as I_x/I. Since the moments of inertia of the vehicle may be carefully adjusted prior to flight, only errors in θ and γ due to $\Delta\omega_x$ and Δt_B will be considered. Thus

$$\Delta\theta_{\Delta\omega_x} = \Delta A_{R_x} \left[\cos (R_x + \mu) t_B - 1\right] - A_{R_x} \sin (R_x + \mu) t_B \Delta(R_x + \mu)$$

$$+ \Delta A_{\omega_x} \left[1 - \cos (\omega_x + \mu) t_B\right] + A_{\omega_x} \sin (\omega_x + \mu) t_B \Delta(\omega_x + \mu) \qquad (32)$$

However, for a tuned system, $\Delta\theta_{\Delta\omega_x}$ is zero for small errors in ω_x. Similarly, $\Delta\theta_{\Delta t_B}$ is zero. In the case of the yaw angle, γ, the following errors would be expected:

$$\Delta\gamma_{\Delta\omega_x} \cong A_{R_x} \left(\frac{I_x}{I} \Delta\omega_x t_B\right) - A_{\omega_x} \Delta\omega_x t_B \qquad (33)$$

and

$$\Delta\gamma_{\Delta t_B} \cong A_{R_x} (R_x + \mu) \Delta t_B - A_{\omega_x} (\omega_x + \mu) \Delta t_B \qquad (34)$$

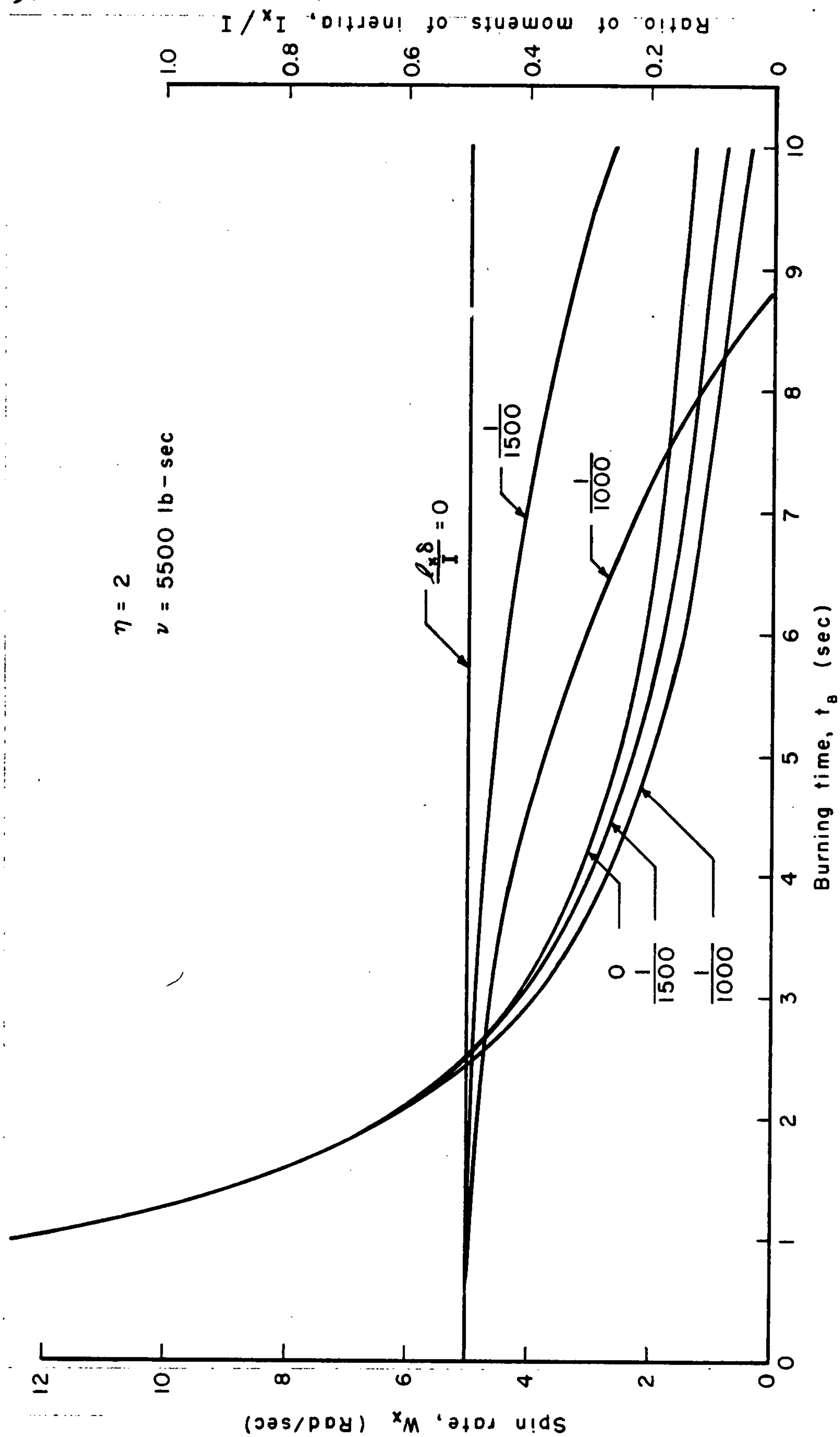


Fig. C-4—Required interrelationship of vehicle parameters which result in a period of oscillation of t_B

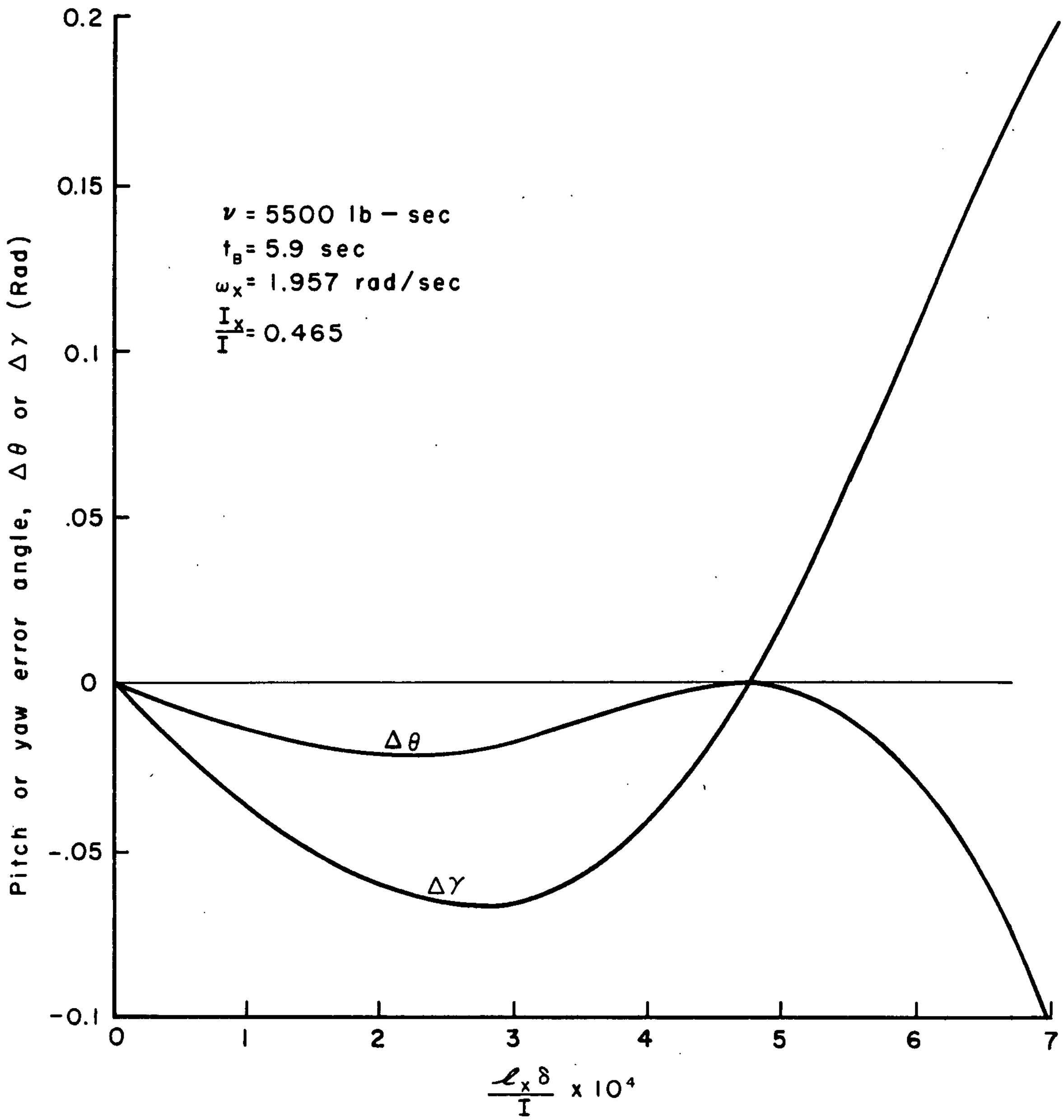


Fig. C-5—Pitch and yaw error angles at the end of burning as a function of the thrust misalignment errors

From the definitions of A_{R_x} , A_{ω_x} and μ upon page 8, it is apparent that $\Delta\gamma_{\Delta t_B}$ is also zero, while $\Delta\gamma_{\Delta\omega_x}$ is approximately zero.

After the burning period is over, Eq. (13) and Eq. (14) give the body rates, ω_y and ω_z . However, from the definition of R_z, and from Eqs. (26) and (27), it can be seen that $R_z\omega_x t_B$ is equal to 2π radians. Thus, ω_y and ω_z are zero for $t > t_B$.

Of course, the most direct approach in reducing the magnitudes of the forced errors in θ and γ is to increase ω_x. For large values of the spin rate, the amplitude constants, A_{R_x} and A_{ω_x} , approach zero, and it is not necessary to adjust the periods of the oscillations.

However, system operation dictates a spin rate of approximately 2 radians per second. Thus, unless the required increment in speed is, by design, kept small so that, in turn, the misalignment torque is small, the periods of oscillation should be tuned to ensure small errors after the propulsion period is over.

The next effect to be examined is that of residual aerodynamic forces. From consideration of Figs. 1 and 2 in conjunction with Eqs. (1) and (2), the following equations may be written:

$$\ddot{\alpha} + \ddot{\theta}_V - \frac{\ell_{cp}}{I}(L_\alpha + D\alpha) + (\dot{\beta} + \dot{\gamma}_V)(\dot{\psi} - R_z\,\omega_x) = 0 \qquad (35)$$

$$\ddot{\beta} + \ddot{\gamma}_V - \frac{\ell_{cp}}{I}(L_\beta + D\beta) - (\dot{\alpha} + \dot{\theta}_V)(\dot{\psi} - R_z\omega_x) = 0 \qquad (36)$$

where L_α is the lift force due to α, acting perpendicular to V,

L_β is the lift force due to β, acting perpendicular to V, and

D is the drag force acting along V. In obtaining Eqs. (35) and (36), all angles except ψ are assumed to be 'small angles.' Thus,

$$L_\alpha = \frac{1}{2} \rho V^2 A_{ref} \frac{\partial C_L}{\partial \alpha} \alpha \quad \text{(a)}$$

$$L_\beta = \frac{1}{2} \rho V^2 A_{ref} \frac{\partial C_L}{\partial \beta} \beta \quad \text{(b)} \quad (37)$$

$$D = \frac{1}{2} \rho V^2 A_{ref} C_D \quad \text{(c)}$$

Upon adopting a circular orbit, and letting $\dot{\psi} = \omega_x$, Eq. (35) and Eq. (36) have the following form:

$$\ddot{\alpha} - \frac{\ell_{cp}}{I} \left(\frac{1}{2} \rho V^2 A_{ref}\right) \left(C_D + \frac{\partial C_L}{\partial \alpha}\right) \alpha + (\dot{\beta} R_x) = 0 \quad (38)$$

$$\ddot{\beta} - \frac{\ell_{cp}}{I} \left(\frac{1}{2} \rho V^2 A_{ref}\right) \left(C_D + \frac{\partial C_L}{\partial \beta}\right) \beta - (\dot{\alpha} R_x) = - \frac{g_o r_E^2}{V r^2} R_x \quad (39)$$

The solutions of Eqs. (38) and (39) may be readily obtained.

$$\alpha = \frac{+ \frac{g_o r_E^2}{V r^2} R_x z_3}{z_1 (z_1^2 - z_3^2)} \sin z_1 t + \frac{\frac{g_o r_E^2}{V r^2} R_x z_1}{z_3 (z_1^2 - z_3^2)} \sin z_3 t \quad (40)$$

$$\beta = - \frac{g_o r_E^2 R_x}{V r^2 \omega_n^2} - \frac{g_o r_E^2 R_x z_3^2}{V r^2 \omega_n^2 (z_1^2 - z_3^2)} \cos z_1 t + \frac{g_o r_E^2 R_x z_1^2 \cos z_3 t}{V r^2 \omega_n^2 (z_1^2 - z_3^2)} \quad (41)$$

where

$$z_1 = \sqrt{\omega_n^2 + \frac{R_x^2}{4}} + \frac{R_x}{2}$$

$$z_3 = \sqrt{\omega_n^2 + \frac{R_x^2}{4}} - \frac{R_x}{2}$$

$$\omega_n^2 = - \frac{\ell_{cp}}{I} \left(\frac{1}{2} \rho V^2 A_{ref}\right) \left(C_D + \frac{\partial C_L}{\partial \alpha}\right)$$

As the aerodynamic effects become negligible, the parameter z_3 approaches zero, while z_1 approaches R_x. If the satellite has a two day operational capability, then the maximum value of t is of the order of 1.75×10^5 seconds. For the angle $z_3 t$ to remain small over this period of time, it is necessary that z_3 have a magnitude not greater than 10^{-6}. Thus

$$\omega_n^2 \cong 10^{-6} R_x \qquad (42)$$

Figure 6 is a plot of $\frac{\ell_{cp}}{I}$, the ratio of the static margin to the pitch-yaw moment of inertia, as a function of altitude with the restriction imposed by Eq. (42).

As a typical example, for an orbit altitude of 800,000 feet with R_x equal to one and with I equal to 20 slug-ft^2, the center of mass and the center of pressure may have a maximum separation of 0.575 inches to ensure that torques due to aerodynamic effects are negligible. With $z_3 t$ a small angle, Eq. (40) and Eq. (41) have the following form

$$\alpha = + \frac{g_o r_E^2}{V r^2} \left[t + \frac{\omega_n^2}{R_x^3} \sin R_x t \right] \qquad (43)$$

$$\beta = - \frac{g_o r_E^2}{V r^2} \frac{R_x}{\omega_n^2} \left[\frac{\omega_n^4}{R_x^4} \cos R_x t \right] \qquad (44)$$

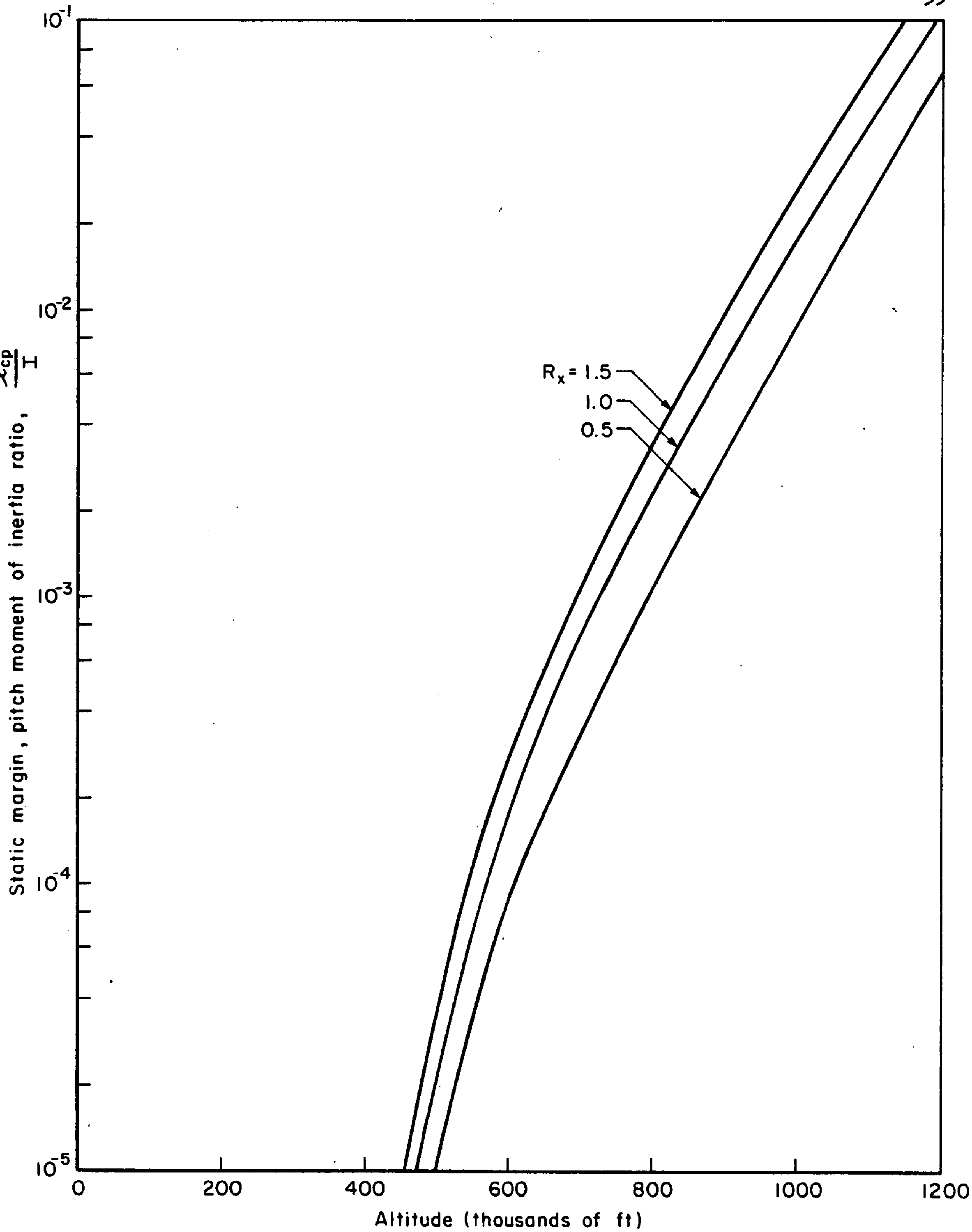


Fig. C-6—Permissible static margin as a function of altitude

Thus α is approximately equal to $\dot{\theta}_{ss}t$, while β is approximately zero, which is the result indicated by Eqs. (7) and (8) for the case of zero initial errors.

Disturbances which arise due to the motion of internal components may be minimized by the proper placement of such gear. As an example, any rotating machinery should be placed, if possible, with the axes of rotation coincident with the vehicle's longitudinal spin axis. The motion of such equipment will then cause perturbations in the spin rate, but will not exert moments which would result in a pitch or yaw motion.

If it were required the spin rate could be regulated by the use of a reaction wheel.

Other disturbances such as those due to meteor impact and magnetic field interactions may occur. However, the former event appears to be very unlikely,[1] while a preliminary examination of the latter effect indicates that for the vehicle under consideration, no significant errors should develop during a two day operational period.

Thus, the expressions for the total error in θ and γ are:

$$\Delta\theta = \pm \epsilon_{\theta} \pm 0.707 \ \Delta A_{\theta,\gamma} \pm \Delta\theta_{\delta} \tag{45}$$

$$\Delta\gamma = \pm \epsilon_{\gamma} \pm 0.707 \ \Delta A_{\theta,\gamma} \pm \Delta\gamma_{\delta} \pm \Delta\gamma_{\Delta\omega_x} \tag{46}$$

In order to evaluate the root sum squared values of $\Delta\theta$ and $\Delta\gamma$, the following values have been assumed:

$$\epsilon_{\theta} = \epsilon_{\gamma} = \pm 0.035 \text{ rad}$$

$$\epsilon_{\dot{\theta}} = \epsilon_{\dot{\gamma}} = \pm 0.035 \text{ rad/sec}$$

$$\omega_x = 1.957 \text{ rad/sec}$$

$$\Delta\omega_x = \pm 0.035 \text{ rad/sec}$$

$$\frac{I_x}{I} = 0.465$$

$$t_B = 5.9 \text{ secs}$$

The preceding values, in conjunction with Figs. 3 and 5 and Eqs. (45) and (46), yield:

$$\Delta\theta_{rss} = 0.087 \text{ rad}$$

$$\Delta\gamma_{rss} = 0.109 \text{ rad}$$

The primary source of rate disturbances is the initial residual error rates. Thus

$$\Delta\dot{\theta} = \Delta\dot{\gamma} = 0.049 \text{ rad/sec}$$

From the point of view of the operational requirements, the attitude stabilization of the vehicle poses no particular difficulties. The assumed performance of the orientation control system is within current capabilities.

(1.) Buchheim, R. W., 'Lunar Instrument Carrier--Attitude Stabilization,' The RAND Corporation, Research Memorandum RM-1730, June 4, 1956 (Unclassified).

Appendix D

GUIDANCE ACCURACY REQUIREMENTS

J. H. Huntzicker

The inaccuracies that may be tolerated in the ascent guidance system can be related to the design altitude, which is determined by a compromise of the photo interpreter's desire for large scale and the requirement to keep aerodynamic forces negligible during the one-day operation. Photographic quality will in general be good, independent of the altitude; however, the intelligence value of the pictures will decrease if the altitude is too great. In this connection, one problem which should be resolved during the satellite test program is the determination of air density as a function of altitude; with these data a minimum operation altitude can be established.

For purposes of determining guidance tolerances a design altitude of 750,000 ft (142 mi) has been selected, with permissible variation being ±250,000 ft (47.3 mi). At these altitudes aerodynamic effects are nominal. If the variation in altitude were to be considerably reduced the design altitude could, of course, be lowered.

This acceptable range of altitude (142 ±47.3 mi) can be interpreted in terms of ascent guidance requirements in the following manner: The minimum or perigee altitude (h_p) and the maximum or apogee altitude (h_a) are uniquely determined by these conditions at the end of final, or third-stage, burning; the magnitude of the velocity (v_3), the direction of the velocity vector with respect to the horizontal (γ_3) and the altitude (h_3). They are related by the following equations:

$$r_a = \frac{v_3^2 r_3^2 \cos^2 \gamma_3}{\mu(1-e)} \tag{1}$$

$$r_p = \frac{v_3^2 r_3^2 \cos^2 \gamma_3}{\mu(1+e)} \tag{2}$$

where

$r_i = h_i + r_e$ (r_e = radius of the earth)

$\mu = g\, r_e^2$ (g = the gravitational constant)

and the orbital eccentricity e is defined by

$$e = \frac{r_a - r_p}{r_a + r_p}$$

and $$e = \left\{ 1 - (2 - v_3^2 r_3/\mu) \frac{v_3^2 r_3 \cos^2 \gamma_3}{\mu} \right\}^{1/2} \tag{3}$$

Figure D-1 presents h_a and h_p as functions of Δv_3 for varying $\Delta \gamma_3$ where $\Delta v_3 = v_3 - v_c$ (v_c = circular velocity at 142 mi) and $\Delta \gamma_3 = \gamma_3$ (i.e., $\gamma = 0$ for a circular orbit). This figure does not include the effect of errors in the altitude of final burning (Δh_3). This effect can be approximated by

$$\frac{dh_a}{dh_3} = \frac{dh_p}{dh_3} = +2 \tag{4}$$

Figure D-1 will yield a measure of the inaccuracies which are tolerable in the control of the ascent trajectory, but this only indirectly. For D-1 to be meaningful in terms of the performance required of the guidance equipment it is necessary to examine the flight path to determine where errors originate and how they propagate to the point of final thrust cut-off (Point 3).

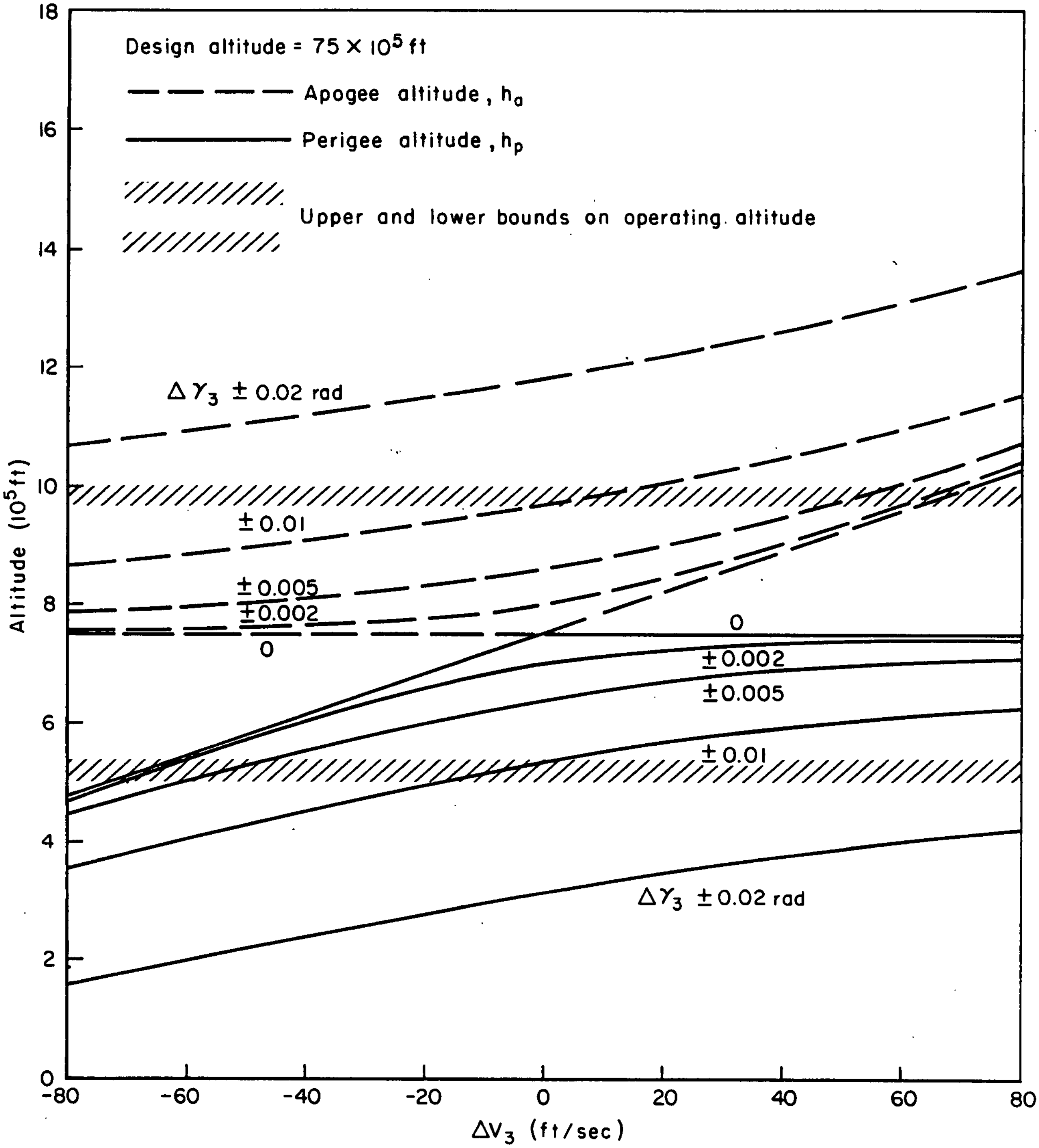


Fig. D-1—Apogee and perigee altitudes related to errors in the magnitude (ΔV_3) and direction ($\Delta\gamma_3$) of final burnout velocity

Figure D-2 illustrates a representative ascent trajectory. At Point 1 second stage burn-out occurs, followed by orientation and spinning of the third stage. A long coasting period (about 4800 n mi) takes the vehicle from Point 1 to Point 2 where third-stage burning occurs. Point 3 is the point of final burn-out and hence the initiation of orbiting.

Inaccuracies in guidance prior to Point 1 will result in errors in altitude, velocity, and direction at Point 1. Each of these errors contributes to errors in all three quantities at Point 3. Errors are also introduced by inaccuracy in the control of the magnitude and direction of the velocity increment added between Points 2 and 3. Due to the small amount of velocity added in the third stage (approximately 300 to 500 ft/sec) the effects of these errors should also be small (e.g., if the velocity varied by ± 2 per cent and the direction by ± 5 degrees the resultant errors in final cut-off conditions are ± 10 ft/sec and ± 1.7 mils, which, if added with zero correlation to the errors originating at Point 1, will be negligible.)

The assumption was made that final burning occurs at a constant range angle measured from Point 1. If final burning were controlled with a preset clock, further errors would be introduced by inaccuracies in the clock and by variations in the time required to reach the desired range angle. The latter introduces an error in range angle of less than 10 miles. Comparable accuracy in the time base would require a clock accurate to about 0.1 per cent ($\sim$80 sec/day). These effects will be insignificant relative to those caused by errors in second-stage cut-off conditions.

On the basis of the preceding considerations the remainder of this study was restricted to an examination of the propagation of errors from

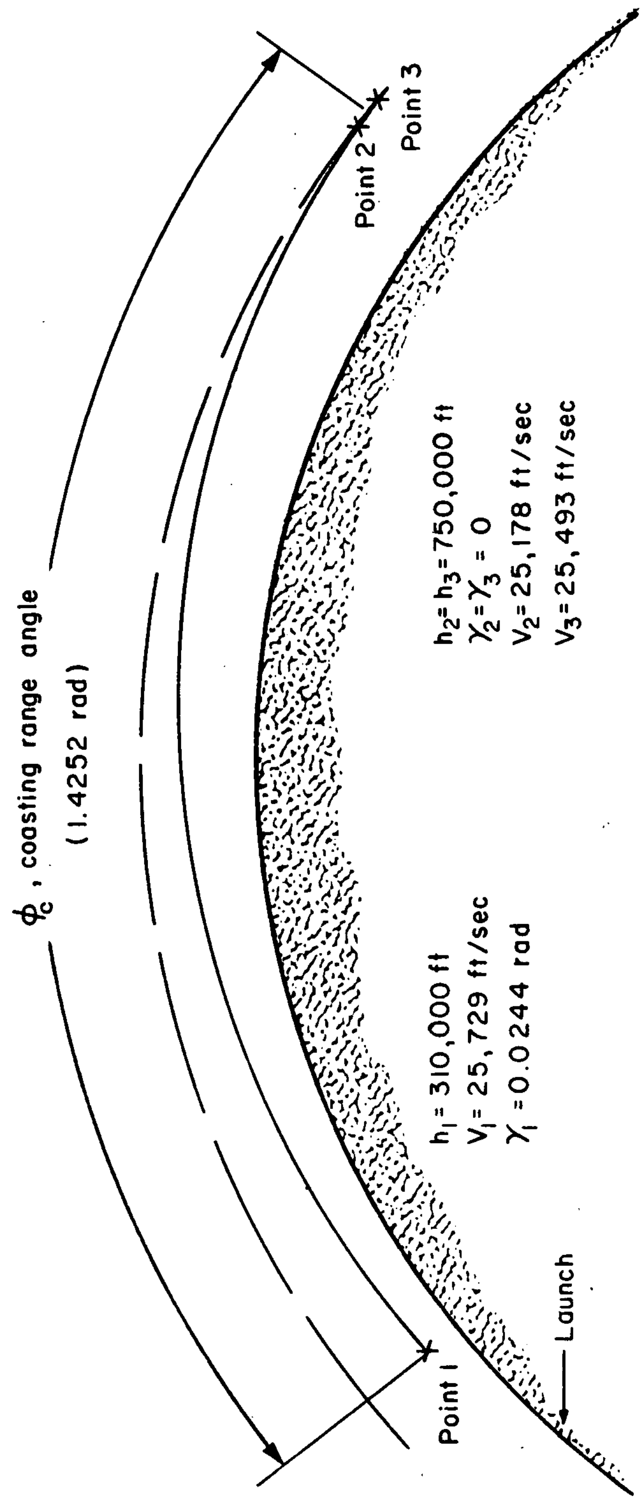


Fig. D-2 — Representative ascent trajectory

Point 1 to Point 3 (nearly identical in space to Point 2). During the coasting from Point 1 to Point 2 the vehicle follows an ellipse described by

$$r = \frac{v_1^2 r_1^2 \cos^2 \gamma_1}{\mu(1-e \cos \phi)} \qquad (5)$$

where ϕ is the range angle measured from the apogee of the ellipse and

$$v_1 = v_{d1} + \Delta v_1$$

$$r_1 = r_{d1} + \Delta r_1$$

$$\gamma_1 = \gamma_{d1} + \Delta \gamma_1$$

The symbols v_{d1}, r_{d1} and γ_{d1} are the design conditions at Point 1 as indicated on Fig. D-2. The values Δv_3, Δr_3 and $\Delta \gamma_3$ are calculated at a point down-range ϕ_c radians (ϕ_c is the design coasting range angle - see Fig. D-2); ϕ_3 will be equal to $\phi_1 + \phi_c$ (ϕ_1 being negative inasmuch as the apogee is down-range from Point 1).

In addition to Eqs. (3) and (4) the following are used:

$$\frac{2\mu}{r} - v^2 = \frac{2\mu}{r_1} - v_1^2 \qquad (6)$$

$$\tan \gamma = \frac{-e \sin \phi}{1-e \cos \phi} \qquad (7)$$

The final errors are then

$$\Delta v_3 = v_2 - v_{d2} \qquad (8)$$

$$\Delta r_3 = r_2 - r_{d2} \qquad (9)$$

$$\Delta \gamma_3 = \gamma_2 \qquad (10)$$

These were calculated for all combinations of a range of values of Δv_1, Δr_1 and $\Delta \gamma_1$. Representative results are plotted as Figs. D-3, D-4 and D-5. As can be seen, the relationships are quite linear within the range of interest and can be approximated by

$$\Delta v_3 = -2.5 \times 10^4 \Delta \gamma_1 - 0.75 \Delta v_1 - 10^{-3} \Delta r_1 \qquad (11)$$

$$\Delta r_3 = 2.1 \times 10^7 \Delta \gamma_1 + 1.5 \times 10^3 \Delta v_1 + 2 \Delta r_1 \qquad (12)$$

$$\Delta \gamma_3 = 0.1 \Delta \gamma_1 + 7.5 \times 10^{-5} \Delta v_1 - 5 \times 10^{-8} \Delta r_1 \qquad (13)$$

A marked negative correlation exists between Δv_3 and Δr_3, the net effect being to limit the variation of h_a and h_p of the final orbit (see Fig. D-1). This is further demonstrated by the inclusion of Δv^* on Figs. D-3, D-4 and D-5, where Δv^* is the velocity error defined with respect to the actual altitude at Point 3 rather than the design altitude. The value of Δv^* is in general only a fraction of Δv_3.

Based on the foregoing we can stipulate the approximate accuracy required of the first- and second-stage guidance equipment. In order to establish an orbit which will remain within altitude limitations of 142 ± 47 mi we could tolerate probable errors in velocity of as much as ±50 to ±75 ft/sec and probable errors in angle of up to ±4 to ±6 mils.

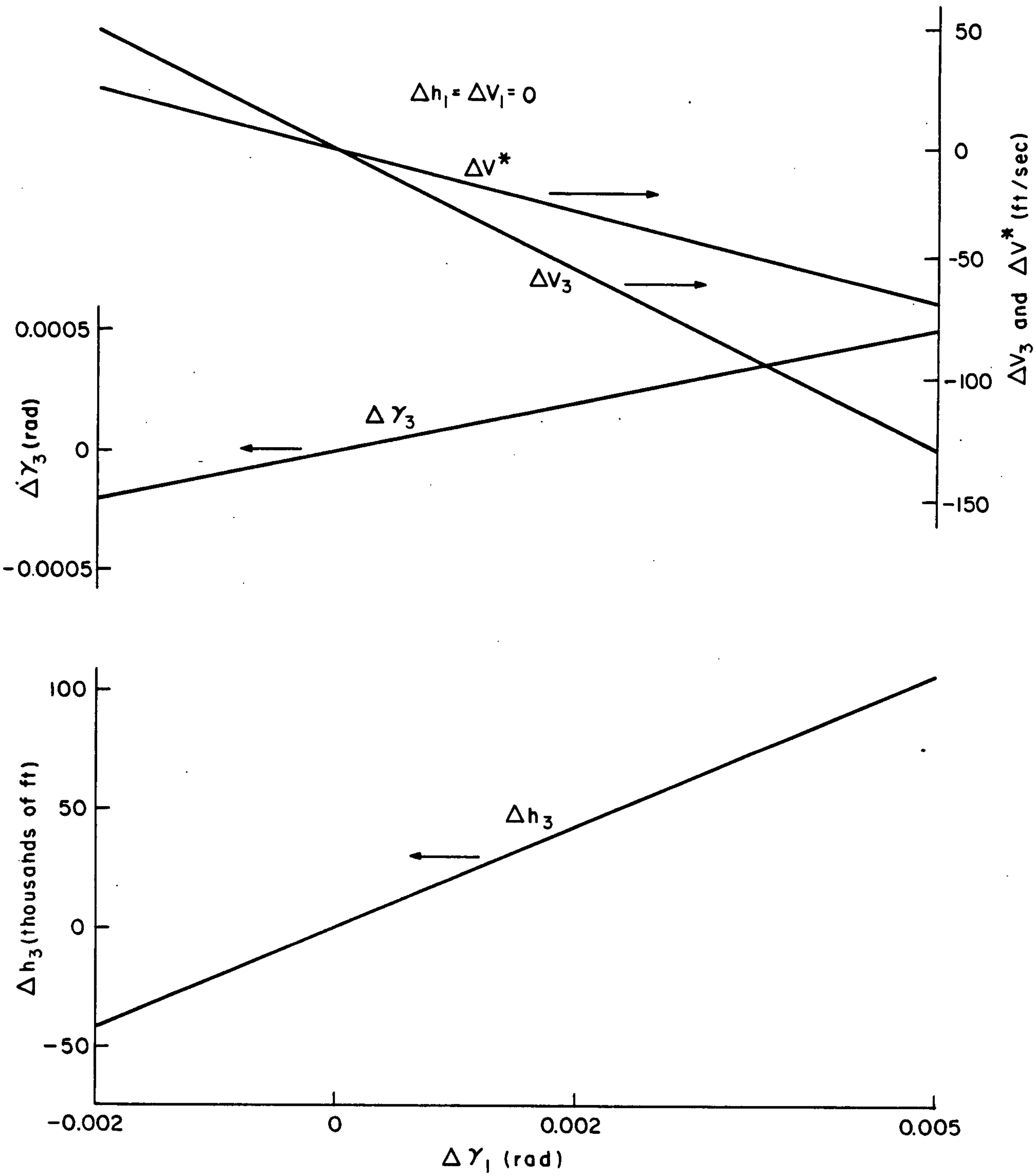


Fig. D-3— Errors in altitude (Δh), velocity (ΔV), and flight path angle ($\Delta \gamma$) at point 3 due to errors in flight path angle at point 1 ($\Delta \gamma_1$)

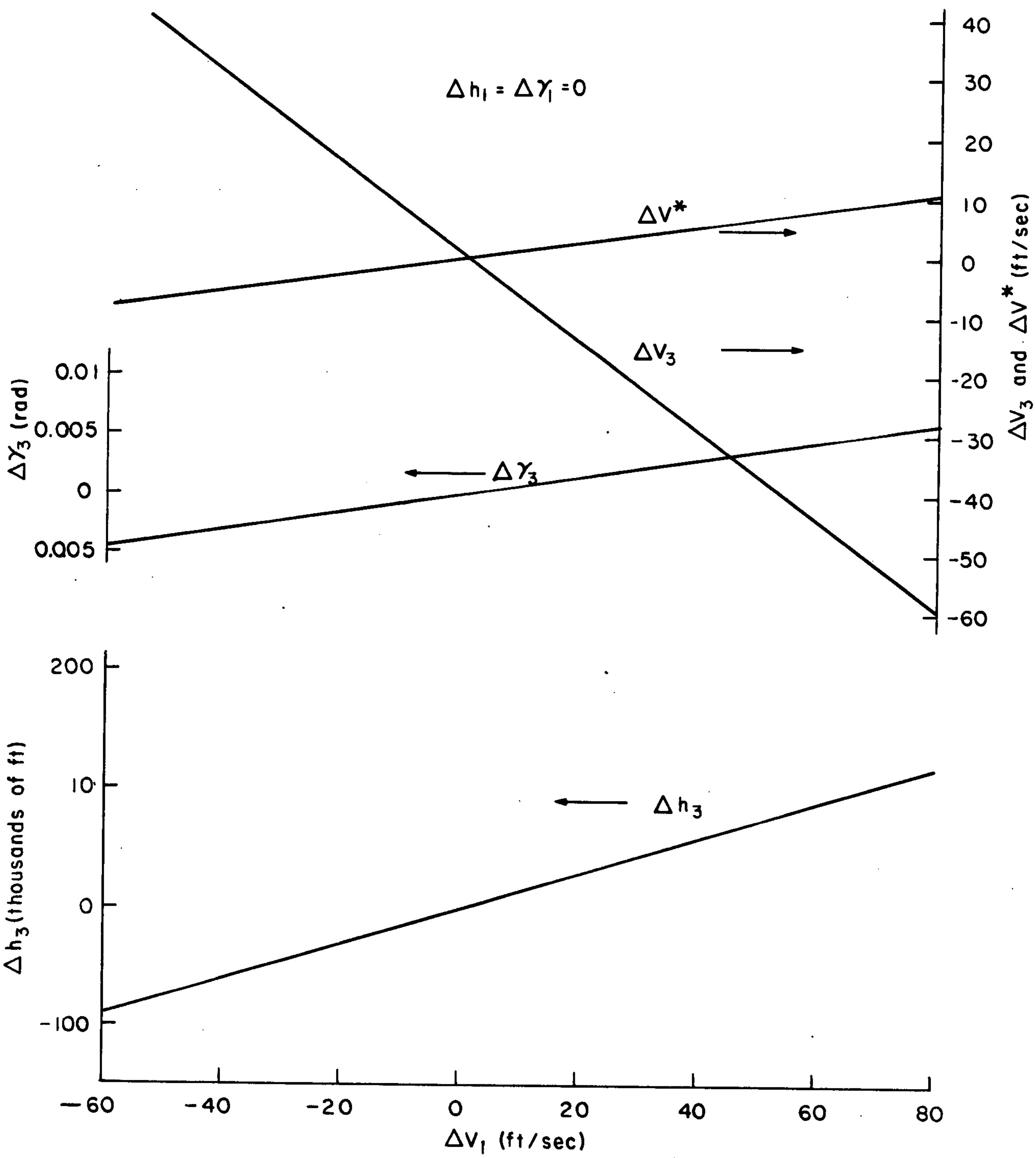


Fig. D-4— Errors at point 3 (ΔV, Δh, ΔV^*, and $\Delta \gamma_3$) caused by errors in velocity at point 1 (ΔV_1)

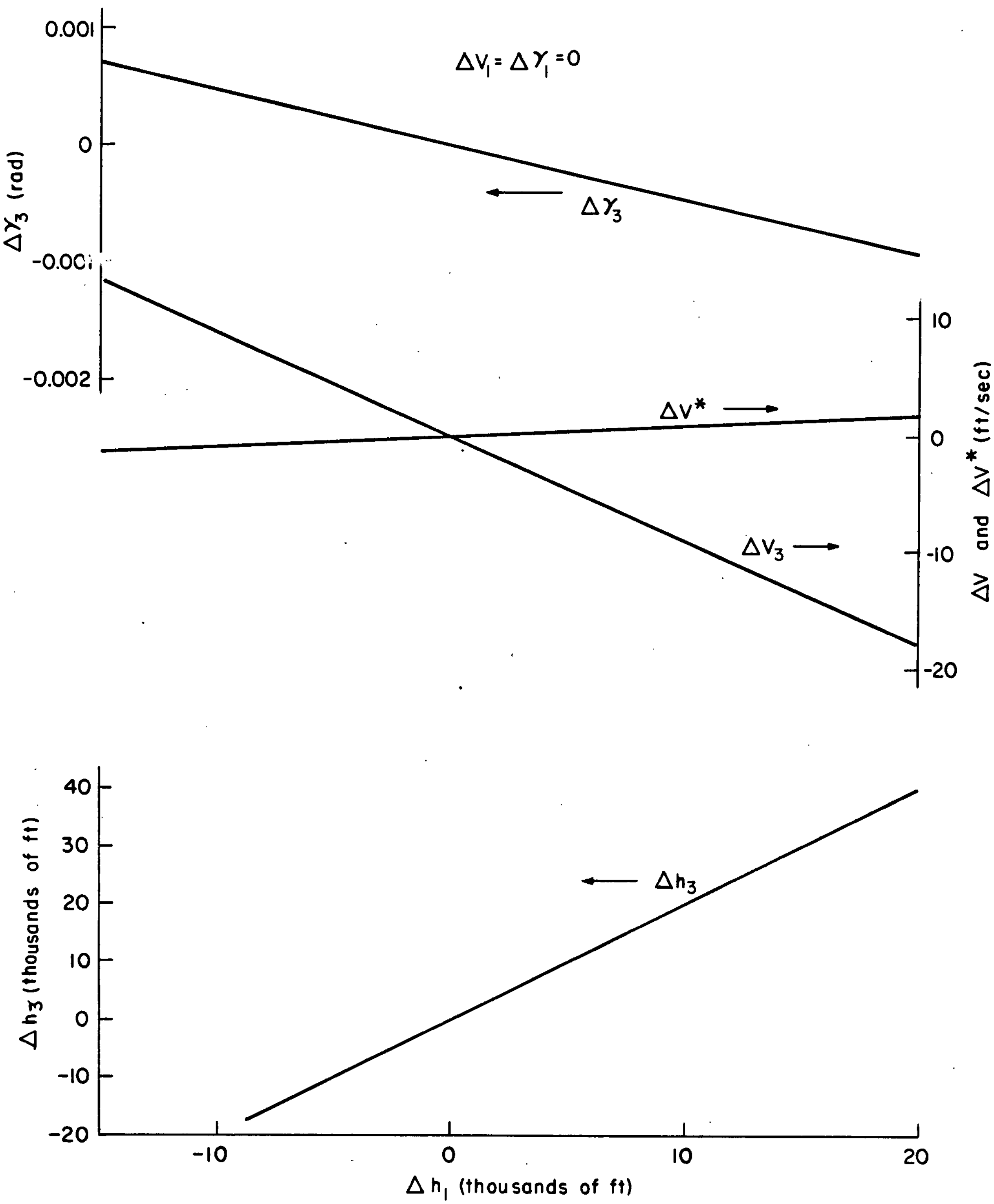


Fig. D-5 — Errors at point 3 (ΔV, Δh, ΔV^*, and $\Delta\gamma_3$) caused by errors in altitude at point 1 (ΔV_1)

Appendix E

FLIGHT MECHANICS

H. A. Lieske

The main features of the orbit of this proposed reconnaissance system are the inclination of the orbit and the spatial orientation of the satellite vehicle: the orbit is to be in a plane passing through the poles, and the satellite is to be oriented so that its roll axis is horizontal at a latitude of 55 deg — that is, the roll axis is at an angle of 35 deg relative to the equatorial plane. A polar orbit requires that the satellite be launched in a southerly direction from Camp Cooke (latitude 34.5 deg North), and that the roll axis of the satellite be oriented so that it is horizontal at a latitude of 55 deg South. If the final orbital velocity increment is to be added at 55 deg latitude south, the total ascent is approximately 5400 n mi. That is, this satellite ascent path is equivalent to the first half of a shallow ballistic missile trajectory of about 10,000 n mi total range. This ascent path will require the maximum performance capability from the booster combination. An alternate method of establishing the orbit with the correct satellite orientation will be discussed later.

The launching vehicle for the satellite consists of a two-stage booster combination — the Thor and the second stage of Vanguard — plus a small solid propellant rocket of the Vanguard third-stage type to give the final orbital velocity increment. The booster stages are described in Appendix F. The vehicle parameters pertinent to the performance study are repeated in Table 1.

Table 1

VEHICLE PERFORMANCE PARAMETERS*

Thor		Vanguard II	
W_{E_1}	= 13,880 lb	W_E	= 1430 lb
W_{prop_1}	= 97,030 lb	W_{prop}	= 3320 lb
I_o	= 245 sec (sea level)	I	= 278 sec (vacuum)
T	= 150,000 lb	T	= 7500 lb
t_B	= 158.5 sec	t_B	= 123 sec

Final Stage (solid rocket)

I = 245 sec (vacuum)

ν^* = 0.750

*These parameters were used in the trajectory analysis.

The burnout velocity of the booster combination was approximated by use of the theoretical velocity potential of each stage minus the approximate values of the velocity losses due to drag and gravity. The velocity loss due to gravity during the Thor powered stage, flown on a gravity-turn path, is shown in Fig. E-1 as a function of the first-stage-burnout path angle. The velocity loss due to drag for a nominal configuration is also included. The second-stage gravity loss is given in Fig. E-2, also for a gravity-turn path. The curve of first-stage-burnout path angle is included to relate the two figures.

An average specific impulse ratio $\bar{I}/I_o = 1.13$ was used for the first stage. The second stage operates at altitudes above 200,000 ft so that its drag deceleration will be small and is neglected in this analysis. The second stage specific impulse will be constant at the vacuum value of 278 sec.

The approximate burnout altitudes of the first and second stages are given in Fig. E-3 as functions of second-stage path angle. The first- and second-stage burnout ranges are approximately 90 and 450 n mi respectively, for trajectories which are nearly horizontal at second-stage burnout.

The final burnout velocity, at a given path angle, is given by

$$V_{B_2} = - g\bar{I} \log_e (1 - \nu_1) - gI_2 \log_e (1 - \nu_2) - \Delta V_{D_1} - \Delta V_{g_1} - \Delta V_{g_2}$$

where the propellant-gross weight ratios ν_1 and ν_2 are determined from the assumed second-stage payload weight and the data given in Table 1.

The second-stage payload weight is given by

$$W_{pay_2} = \frac{W_{orb}}{1 - \nu/\nu^*} + 25$$

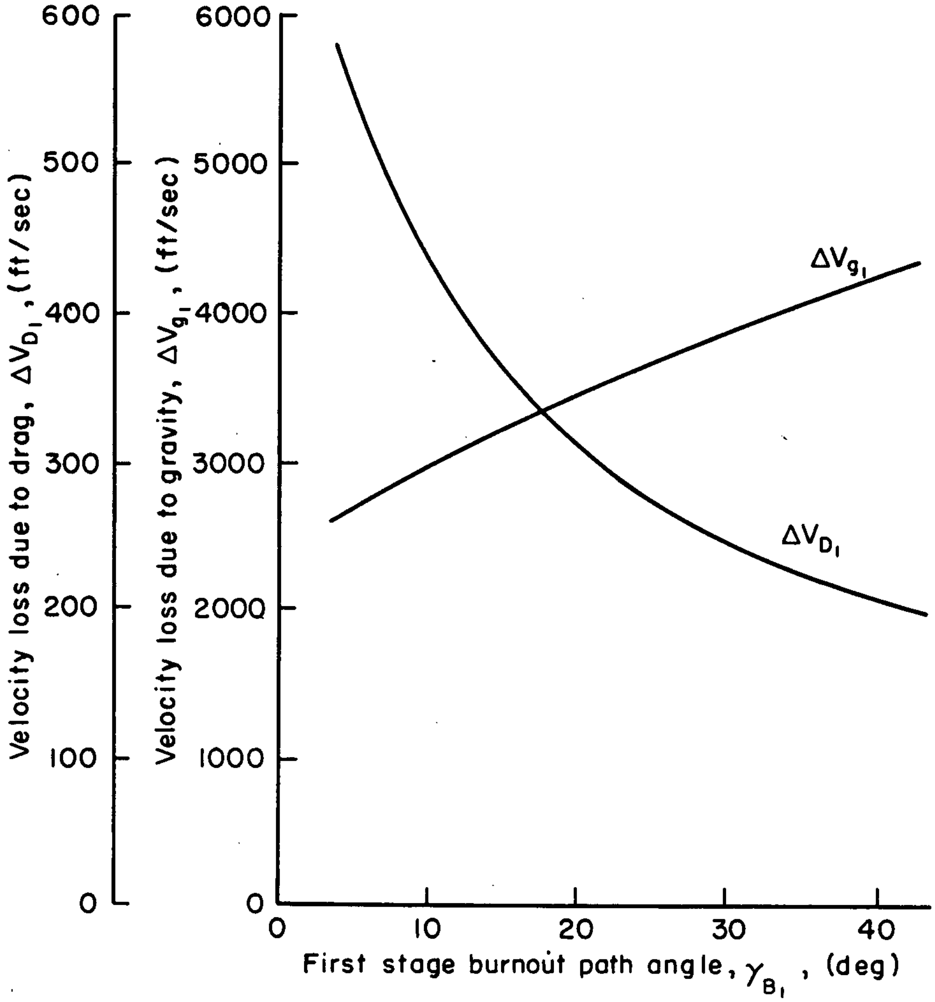


Fig. E-1 — Velocity loss during first stage

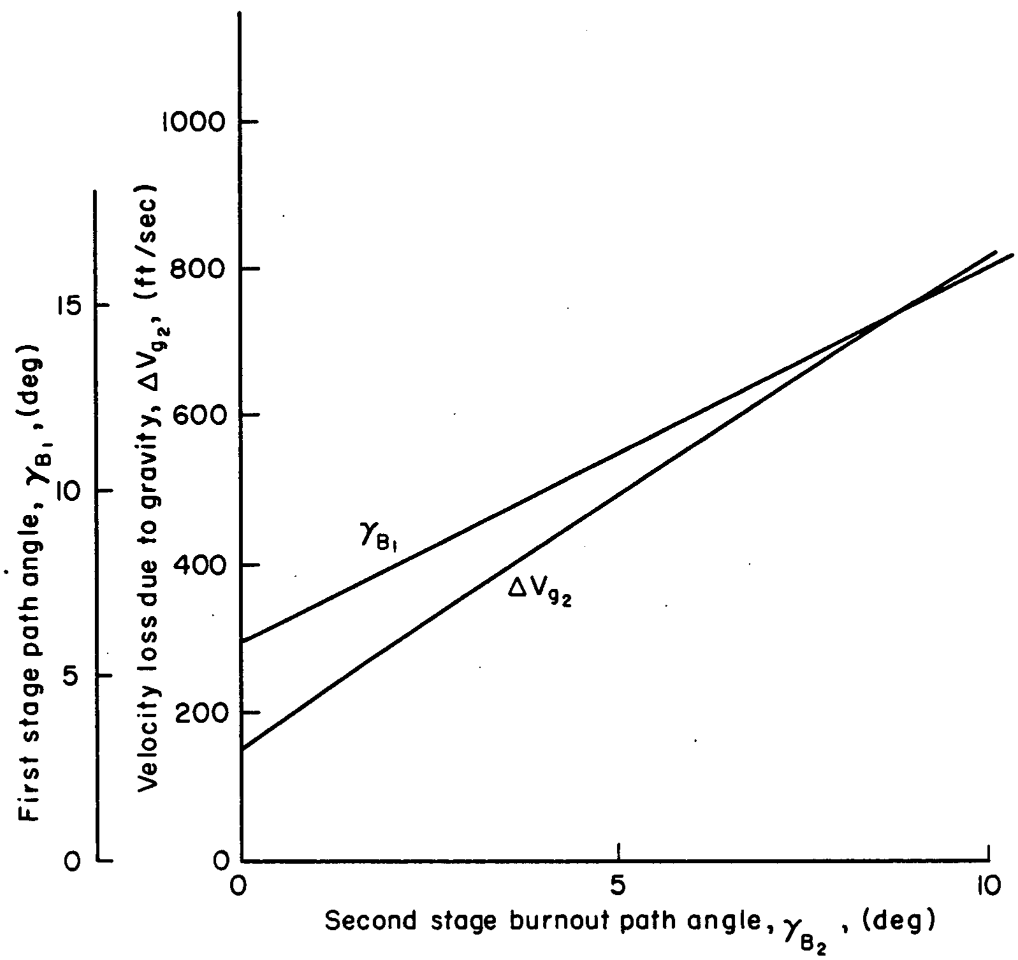


Fig. E-2—Velocity loss during second stage

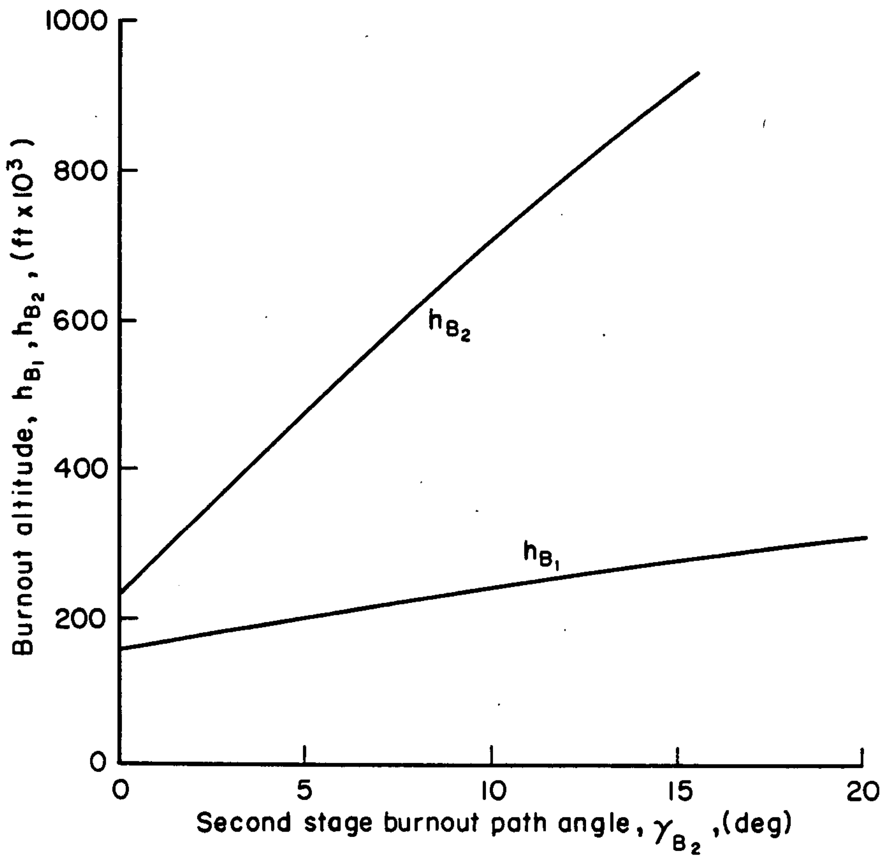


Fig. E-3—Burnout altitudes

with a propellant — total motor weight ratio v^* of approximately 0.75 (see Appendix F), and the required v value computed from the orbital velocity increment using a specific impulse $I_3 = 245$ sec.

The ascent trajectory for an orbiting weight of 300 lb launched from Camp Cooke directly into the orbit at 55 deg south latitude is shown in Fig. E-4. (This ascent trajectory was computed for a circular orbit at an altitude of about 180 stat mi.) The total booster velocity potential, with a second-stage payload weight of 350 lb, is 29,350 ft/sec which is decreased to a burnout velocity of 25,700 ft/sec by the losses due to drag and gravity on this shallow trajectory. The second-stage-burnout path angle is 2 deg at an altitude of 335,000 ft, resulting in an eccentricity of about 0.035 for the free-flight ascent ellipse. The circular orbit velocity increment required for this trajectory is only 450 ft/sec, which is typical of a long range satellite ascent trajectory. The solid rocket motor for this orbital velocity increment will weigh about 25 lb. The total required velocity potential of the three powered stages is 29,800 ft/sec for this ascent trajectory.

The effect of total ascent range on the allowable orbiting weight is shown in Fig. E-5. It can be seen that the allowable orbiting weight for this vehicle combination could be about 400 lb, or an increase of about 30 percent, if the ascent range were about 2700 n mi, or half the range required for launching the satellite from Camp Cooke and entering the orbit at a latitude of 55 deg South.

The relative velocity contributions of the booster combination and of the final stage will vary as the ascent range is changed. As the ascent range is decreased, the required orbital velocity increment is increased while the booster velocity increment is decreased. The net result is an

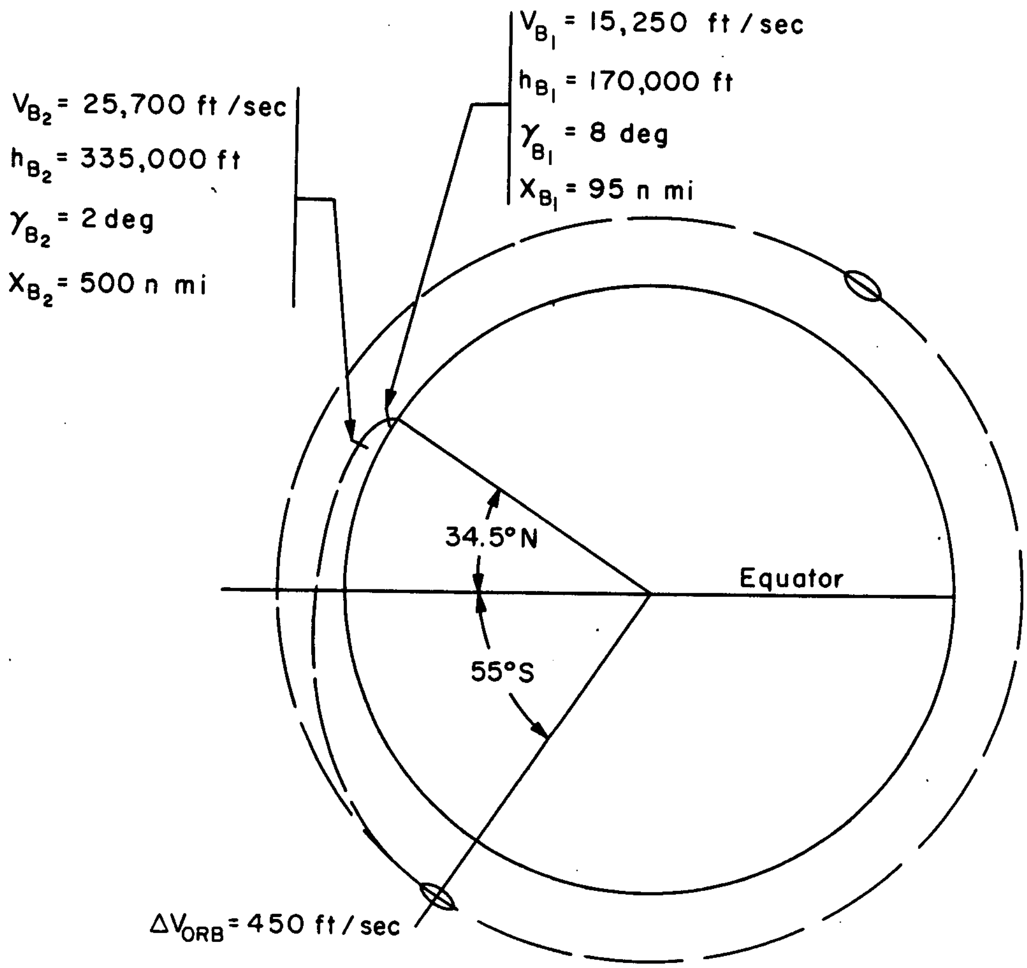


Fig. E-4 — Typical ascent from Camp Cooke

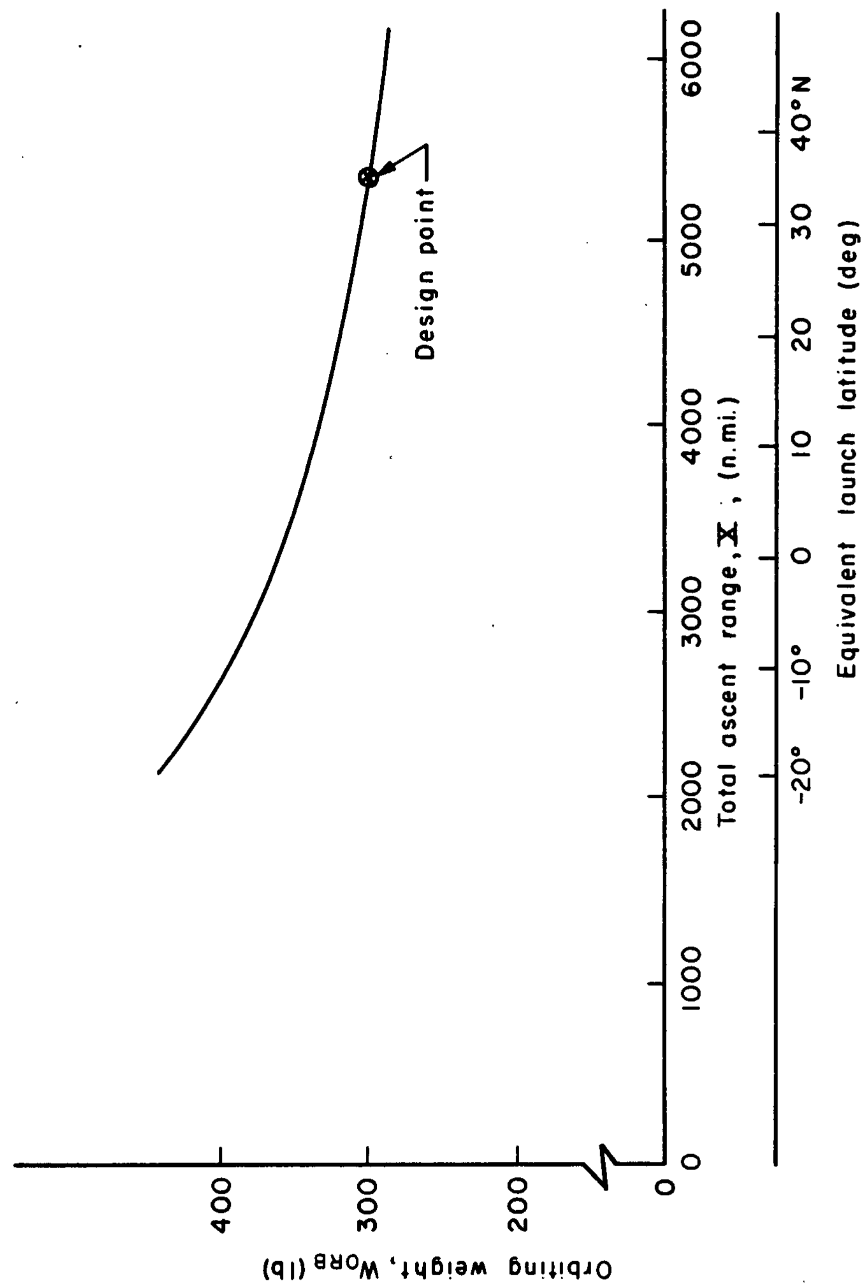


Fig. E-5— Orbiting weight vs ascent range increase in the allowable orbiting weight as the ascent range is reduced to a more nearly optimum value for this booster combination.

The lower horizontal scale of Fig. E-5 indicates the corresponding launch latitude for orbit entry at 55 deg South. For example, if the launch site were at a latitude of approximately 10 deg North (Panama), the allowable orbiting weight for this booster combination would be increased by about 10 per cent. Alternately, if the orbiting weight were fixed at a value of 300 lb, a small performance margin would be available.

Another possible scheme of establishing the satellite in an orbit with the required vehicle orientation (horizontal at 55 deg South) is sketched in Fig. E-6. The trajectory parameters at second-stage burnout are such as to lead to a free-flight ellipse which has an apogee altitude somewhat higher than the required orbital altitude at a shorter distance from the launch site. After second-stage burnout, the satellite vehicle is oriented correctly (roll axis at 35 deg relative to the equator) and spun to the required roll rate. When the vehicle coasts up to the required orbital altitude at an intermediate latitude, l_i, the third stage, which is oriented as mentioned above, adds the appropriate velocity increment to the vehicle's velocity in the ellipse so that the resultant velocity is directed horizontally at that point with a magnitude equal to the required circular orbital velocity. The final velocity increment required for this method of orbit injection will be somewhat larger than the value required for injection at 55 deg South (Fig. E-4), so that the final-stage gross weight will be increased slightly. The variation of orbiting weight with range to apogee shown in Fig. E-5, however, is such that the required weight increase may be compensated for.

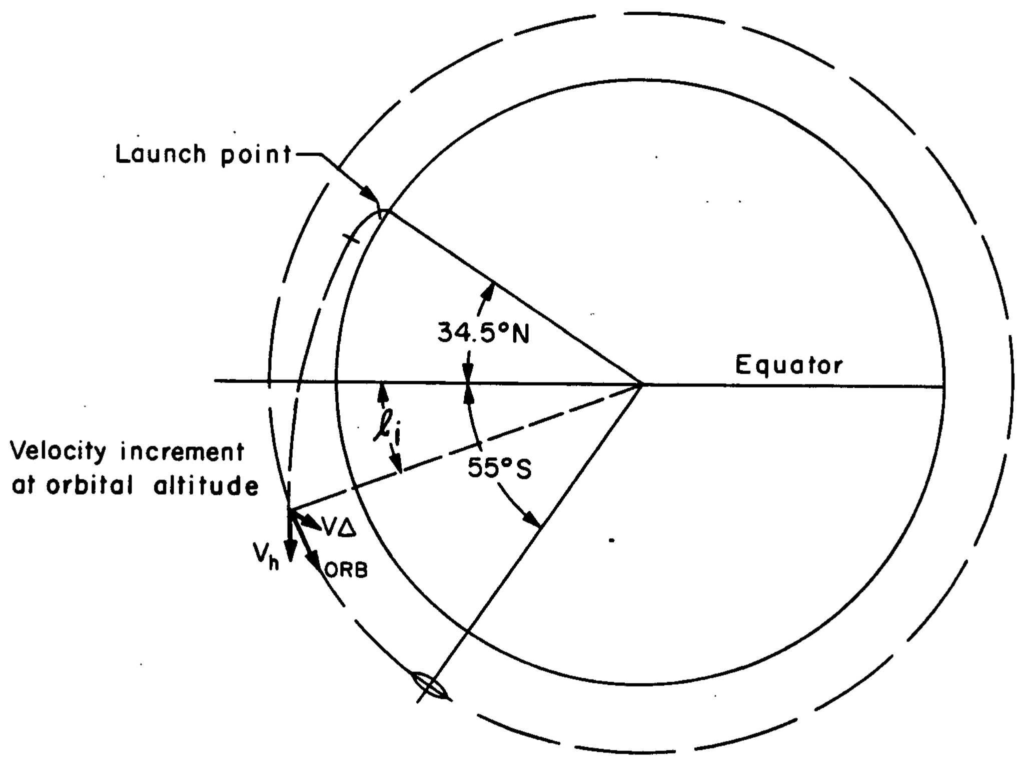


Fig. E-6 — Alternate satellite ascent

Appendix F

VEHICLE DESIGN SUMMARY

E. C. Heffern

FIRST-STAGE BOOSTER

Thor (WS-315) IRBM is used to provide the initial rocket-powered boost for the proposed reconnaissance satellite. Based on a preliminary review of the Thor airframe and its major components and systems, it appears that an upper stage, or stages, weighing 5000 lb could be placed on the Thor without modification to the basic airframe and its primary components. The estimates presented in this report are intended to outline the more basic considerations associated with the use of the Thor as a first-stage satellite booster, and are not intended to reflect a complete analysis.

For the trajectory analysis presented in Appendix E, a weight summary of the Thor missile is given in Table 1. These data are based on information contained in Refs. 1 and 2.

Table 1

WEIGHT SUMMARY - THOR MISSILE (lb)

Structures Group	3060	
Propulsion Group	2380	
Guidance and Control Group*	1565	
Separation System	30	
Electrical System*	200	
Dry missile - less upper stages		7235
Unusable Propellants	1525	
Pressurization Gases	372	
Burn-out Weight - less upper stages		9132
Usable Propellants	97,030	
Take-off Weight - less upper stages		106,162

*Modified weights - see discussion below.

The modified guidance and electrical system weights are due to the omission of the ACSP guidance unit and a portion of the vernier system for the satellite mission. Elimination of the guidance unit reflects a weight reduction in the electrical power system, as well as in the guidance system, because of reduced power requirements. (It is assumed that the required electrical power is obtained from the same type battery-inverter power source used in the early Thor missile.) Since vernier rockets will be required only for control during the main-rocket boost in this application, the vernier propellant tanks may be omitted. The allowance for unused propellants shown in the table includes a 1 per cent reserve for propellant utilization errors.

Although the axial loads resulting from the 5000-lb upper-stage load (the design re-entry-body weight is 3500 lb) are estimated to be within the load-carrying capability of the Thor airframe, the increased length of the upper stages may result in a larger airload and larger bending loads, which may require localized stiffening in the guidance section and in the section between the tanks. Because of the differences in the diameter of the re-entry adaptor ring and the body diameter of the upper stage, it may be desirable to consider a modified nose, or guidance, section for the Thor missile. The actual design of this section would, of course, be determined by the actual design choice of the upper stage. If the second stage of the Vanguard system is used on the Thor booster, this guidance section may be modified as shown in Fig. F-1.

SECOND-STAGE BOOSTER

The second-stage of the Vanguard vehicle is utilized as an example

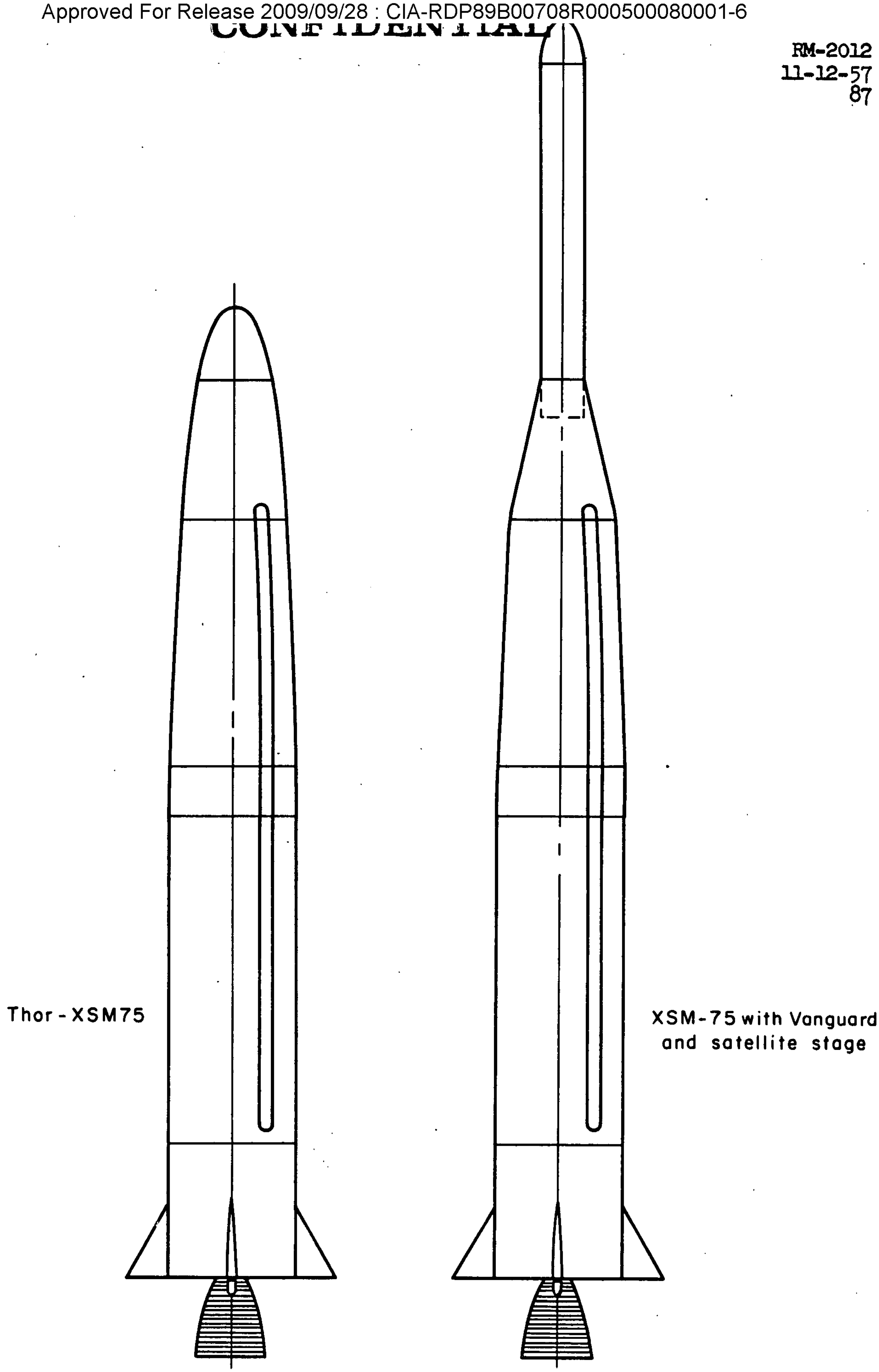


Fig. F-1 — General arrangement

of the type of vehicle, and the performance required, for adaptation to the Thor airframe for satellite missions. Although other missile components could be included for this application, it is believed that the Vanguard second-stage could be available at an earlier date and would have a greater flexibility than other components; for example, the 117L, or a modified Sergeant-type solid-propellant booster with a special control system added for vehicle orientation after burn-out.

The Vanguard second stage consists of a propulsion package manufactured by the Aerojet-General Corporation and assembled into a complete second stage by the Martin Company with the addition of the guidance (autopilot) and control components. The basic ascent trajectory and sequence of operations for the proposed satellite are predicated on many of the features of the Vanguard design. That is, there are two phases of powered flight, followed by a coast period during which the missile is oriented and the third stage is spun around its roll axis prior to third-stage ignition. The weight of the satellite package discussed in the following secion, 300 lb, is less than the weight carried by this section in the Vanguard vehicle (approximately 500 lb).

This summary discussion of the Vanguard second-stage vehicle, and the performance data presented in Table 2, are based on Refs. 3 and 4.

The propulsion package (Aerojet AJ-10) includes a 7500-lb thrust (vacuum conditions) rocket engine using unsymmetrical dimethylhydrazine (UDMH) and inhibited fuming nitric acid (WIFNA) as propellants. The engine is regeneratively cooled and is gimbal-mounted to provide thrust vector control. The propellants are pressure-fed to the thrust chamber by the pressurizing gas (helium augmented by a solid gas generator). The

Table 2

VANGUARD SECOND STAGE

Weight Data

Structure	150	
Powerplant (including tanks) . . .	450	
Controls and guidance	270	
Electrical system	150	
Dry missile weight (less upper stage). . .		1020
Residuals	60	
Burn-out weight (less upper stage)		1080
Usable Propellants	3320	
Gross stage weight		4400

Performance Data (vacuum conditions)

Thrust	7500 lb
Specific impulse . . .	278 sec

pressurization system (helium tank) is located between the propellant tanks. After main-stage burn-out, the residual pressurizing gases are used in the control jets for orientation control during the coasting phase of the trajectory.

The second-stage structure includes an aft skirt, tank section, and the forward instrument and housing section. The aft section is constructed of a magnesium alloy, while the tank section (with the integral pressurization sphere) is heat-treated stainless steel. The forward instrument and housing section are of a magnesium-thorium alloy. This section houses the guidance equipment, the spin assembly, and the third-stage solid propellant rocket. The satellite (orbiting stage) is attached to this section as shown in Fig. F-2.

THIRD-STAGE ROCKET

The long-range ascent path for this type of satellite requires the addition of only a small velocity increment (about 450 ft/sec) to place the vehicle in orbit. A shorter ascent path would permit a heavier payload to be carried, but the increased ΔV required for the orbit would call for a larger unguided solid-propellant rocket than is stipulated here, with attendant undesirable increases in angular and velocity errors (see Appendix E). Uncertainties in the total impulse of the solid rocket (about 1 per cent) could be expected to vary the actual velocity increment by 1 per cent or about 5 ft/sec for the selected trajectory, which can easily be tolerated.

The design of the solid third-stage rocket is based on a shortened Scale-Sergeant case and grain using Vanguard third-stage propellants, which gives 245 sec specific impulse at altitude. The propellant weight

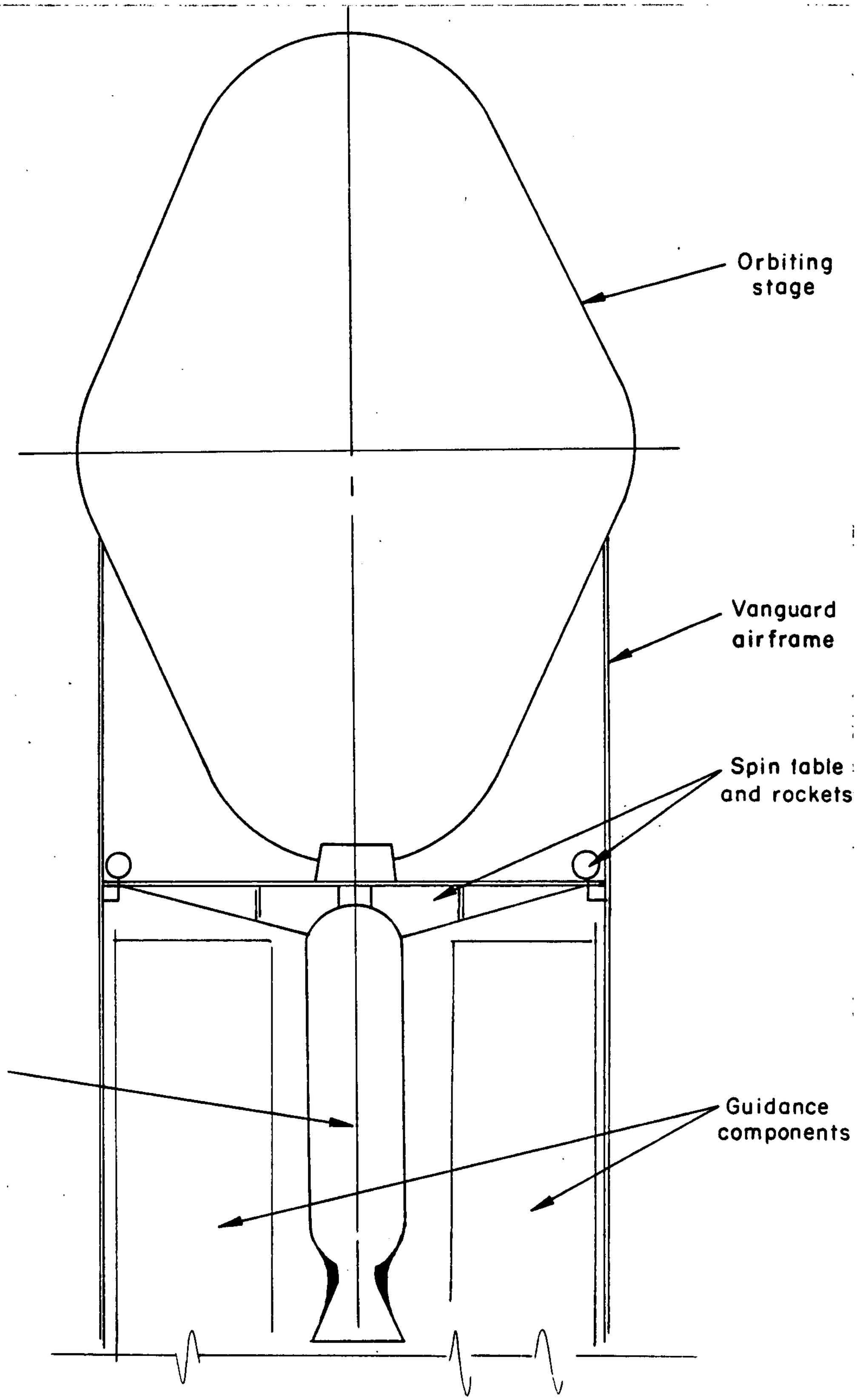


Fig. F-2 — Satellite and third stage rocket of 20 lb with an inert weight of 6 lb will give the required velocity increment of 450 ft/sec.

SATELLITE STAGE

For this discussion, the satellite stage is divided into three major components; the photographic installation, the recovery system, and the structure. The photographic equipment, discussed in Appendix B, is considered to be packaged in a short cylindrical section (35 inches in diameter, 6 inches in length) as shown in Fig. 1 on p. 5 (a hole is provided to allow packaging the rocket case as shown in the figure). A list of the components, and the estimated weights are included in Table 3.

The recovery system includes the tracking and recovery beacon, and the retro-rocket which is the heaviest single component in the satellite. For simplicity it is suggested that the entire satellite be recovered for the 12 inch camera. As larger cameras are incorporated, it may be desirable to recover only the film package, allowing a greater percentage of the satellite to be utilized for useful payload.

For the design proposed in this report, the retro-rocket delivers approximately 1500 ft/sec velocity increment to initiate descent from orbit. The Sacle-Sergeant motor is adequate for this application. Weight of the retro-rocket is about 22 per cent of the total weight of the orbiting stage. Table 4 summarizes the weight breakdown of the recovery system.

The basic configuration of the satellite is a modified double cone with spherical ends as shown in Fig. 2 on p. 6. The forward surface is covered with a plastic-fiberglas material for heat protection through vaporization during re-entry. The shell is constructed of a magnesium allow stiffened with a honey-comb or foam plastic for rigidity. The aft

Table 3

PHOTOGRAPHIC INSTALLATION WEIGHTS (lb)

Camera System		51.5
Lens & lens mount	20	
Film & film spools	12.5	
Film transport	5	
Slit plate	1	
Quartz window	3	
Motor and battery	10	
Altitude Sensor		8.5
Window	0.5	
Sun sensor	1.	
Battery	3.0	
Clock	1.0	
Misc.		
Container		20
Metal parts	15	
Environment	.5	
Total Installation Weight		80

Table 4

RECOVERY SYSTEM WEIGHTS (lb)

Retro-rocket		60
Propellants	51	
Inert parts	9	
Tracking beacon		16
Beacon		
Transponder		
Battery		
Antenna & cable		
Recovery beacon		9
Beacon		
Battery		
Antenna & cable		
Total installation weight		85

portion of the shell structure is covered with a thin plastic coating for heat protection. A summary of the structural weights is given in Table 5.

Table 5

STRUCTURE WEIGHT SUMMARY (lb)

Metal skin	15
Heat shield	65
Aft plastic covering	10
Plastic foam	40
Total structure weight	130

REFERENCES TO APPENDIX F

1. Douglas Aircraft Company, Report SM-27156, Thor WS-315A Technical Report, January-December 1956 (Secret).

2. Douglas Aircraft Company, Report SM-27034, Thor WS-315A Weight Status Report, December 15, 1956 (Secret).

3. U.S. Naval Research Laboratory, Design Specifications for Vanguard Launching Vehicle, NRL Spec. No. 4100-1, revised 29 March 1956 (Confidential).

4. U.S. Naval Research Laboratory, Project Vanguard Report of Progress, Status, and Plans, 1 June 1957 (Confidential).

Appendix G

TRACKING

R. T. Gabler

Tracking the reconnaissance satellite will be required to establish an orbit sufficiently accurate for coordination of photographic data, for triggering the retro-rocket at the desired time for descent, and for establishing the descent path to facilitate impact location. Because of the near-polar orbit, location of tracking stations at high latitudes will be favored. The required number and location of trackers will be dictated partly by the guidance accuracy on which one can depend.

Since tracking accuracy deteriorates at low elevation angles, it is highly important that the satellite pass within a range which permits sufficient tracking data to be obtained at elevation angles greater than 20°. For a nominal 150-mile (statute) height, this requires that the satellite pass within 5° of the station (measured on a great circle path), or within 345 miles ground range. To ensure against guidance inaccuracy in launching, it is suggested that two or three trackers be used in an arrangement which generally places them with a 200 mile east-west separation. For a launching from Camp Cook and recovery in the Pacific, this would indicate Alaska as a favored location for these trackers, from which the descent would be commanded and the impact point predicted. Three such trackers located at a reasonable high latitude in the eastern part of the United States or Canada would be required to track an early pass for orbit prediction.

Tracking data would be in the form of two angles and a range to give position fixes necessary for orbit prediction. A minimum of two positions and the corresponding time are required. Additional tracking data would necessarily be used in reducing the effect of noise errors. The use of range information relaxes considerably the requirement for angular accuracy. To obtain range, a transponder in the satellite is required. This, however, is consistent with the requirement for a command receiver in the satellite for firing the retro-rocket. For a CW system, modulation of a command transmitter at the tracker and remodulation in the satellite in a manner similar to that used by Azusa for range measurement is a feasible scheme.

The frequencies used would be nominally 500 mc for the satellite transmitter and 200 mc for the ground-command transmitter. Since circular polarization is required, the satellite antenna would be a turnstile or helical design so located that the inertial orientation of the vehicle will point the axis of the antenna toward the tracker to within at least 60°. Amplitude modulation of the output stage of the satellite transmitter, which will have a crystal-controlled master oscillator, will permit tracking in angle by either an interferometer-type system or a conical scanning system. It would permit range measurement to be made by phase comparison of the modulation frequency which is transmitted one way on a nominal 200-mc carrier and back on a nominal 500-mc carrier.

For a satellite at 150 (statute) miles height on a circular orbit, a velocity increment of 1500 ft per sec delivered along the longitudinal axis when the vehicle is at 65° N. Lat. (over Alaska) will cause a descent to about 200,000 ft altitude in about 6 minutes with impact about 3.5 minutes later. If commanded at 65° N. Lat. the impact will occur at about 32° N. Lat. Generally, it would be preferred that the satellite pass near the zenith with respect to the tracker since this gives the maximum amount of useful tracking data, but conventional radar trackers are not gimballed to permit tracking through the zenith.

Preliminary estimates indicate that a tracker which can measure angles to about 0.1 mil and range to about 0.1 mi of systematic error, will permit prediction of the impact point to about 3 n mi if the tracking is accomplished after the descent has been commanded. Angle information good to about 0.1 mil (systematic error) can be obtained from either a conical-scan system or an interferometer type.

A CW-type transponder can be designed with smaller space and weight requirement in the vehicle than a pulse system with the required performance. It is estimated that a pulse-type transponder can be adapted from an existing design and will weigh about 15 pounds with power supply. A CW-type could be built to weigh about 10 pounds with power supply.

Since early availability is of paramount importance, one should probably consider adapting an existing tracking radar such as the FPS-16, or Nike missile-tracking radar to do the job. This would require modification of the range circuits, at least in the Nike. If one considers the use of shipborne radars and probably reduced accuracy in prediction, the SPG-49 is a good possibility.

For a recovery beacon it is proposed that a spring-ejected whip antenna be extended vertically through the satellite skin after impact. The antenna will be excited by a 50 mc oscillator designed to radiate about 0.25 watt. The signal strength at 50 miles would be about 35 μv/meter which should be adequate for airborne direction finding. The estimated weight for this design, for a 3-day operation, totals about 5 pounds for the transmitter, antenna, and batteries.

Appendix H

RE-ENTRY AND RECOVERY

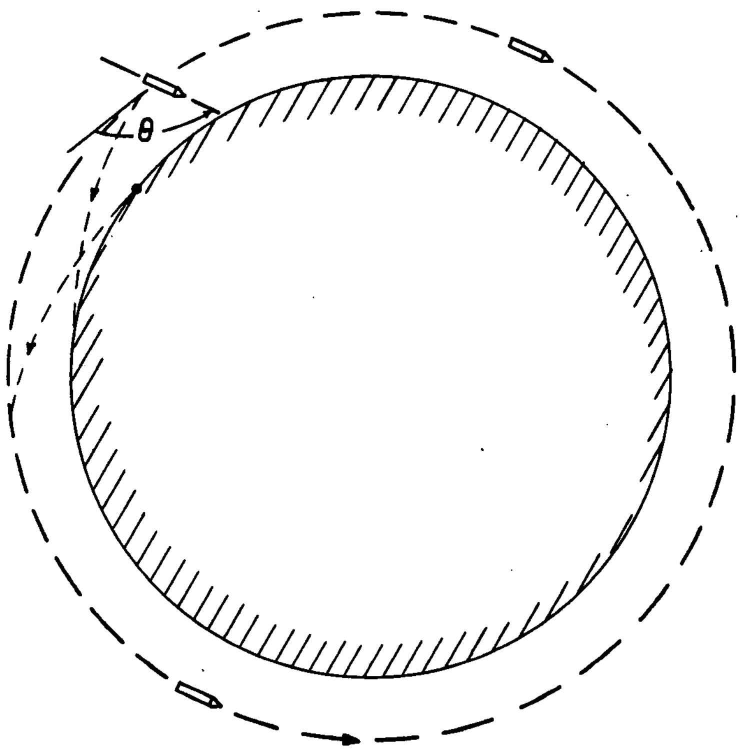
C. Gazley, Jr.

DEPARTURE FROM ORBIT

Descent from the orbit is achieved by the command firing of a braking rocket as described in Ref. 1. Since the vehicle's orientation is conditioned by the geometry necessary for photography of a specific area, and because of the relative positions of that area and the desired recovery area, the braking rocket is fired forward and upward. This results in a downward and backward velocity impulse superimposed on the orbital velocity. The resulting velocity vector is oriented downward at an angle γ with the local horizontal (Fig. H-1). The vehicle is now effectively in a ballistic trajectory comparable to the 'low-angle' (i.e., lower than optimum) path of a long-range ballistic missile.

For the desired range of about 2,000 n mi and an impulse angle $\theta = 110\text{-}125^{\circ}$, a velocity impulse $\Delta V = 1{,}500\text{-}2{,}000$ ft/second is required. This yields a resultant velocity $V_R \approx 24{,}800$ ft/second and a path angle $\gamma = 3\text{-}4^{\circ}$. Descent from this point follows a 'vacuum' path down to an altitude of about 250,000 ft. Below this altitude, atmospheric drag effects increase and ultimately predominate over gravitational influences.* Vacuum ranges are shown in Fig. H-3 as a function of the magnitude and angle of the velocity increment. Descent from the orbital altitude to 250,000 ft is accompanied by a velocity increase to 25,500 ft/second and an angle

*The altitude region where drag becomes important depends, of course, on the drag-mass characteristics of the body.



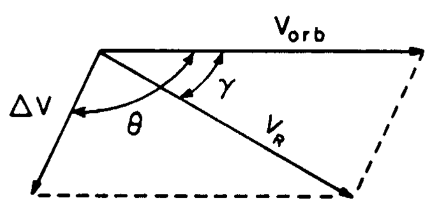


Fig. H-1—Vehicle orientation in orbit and at time of velocity increment

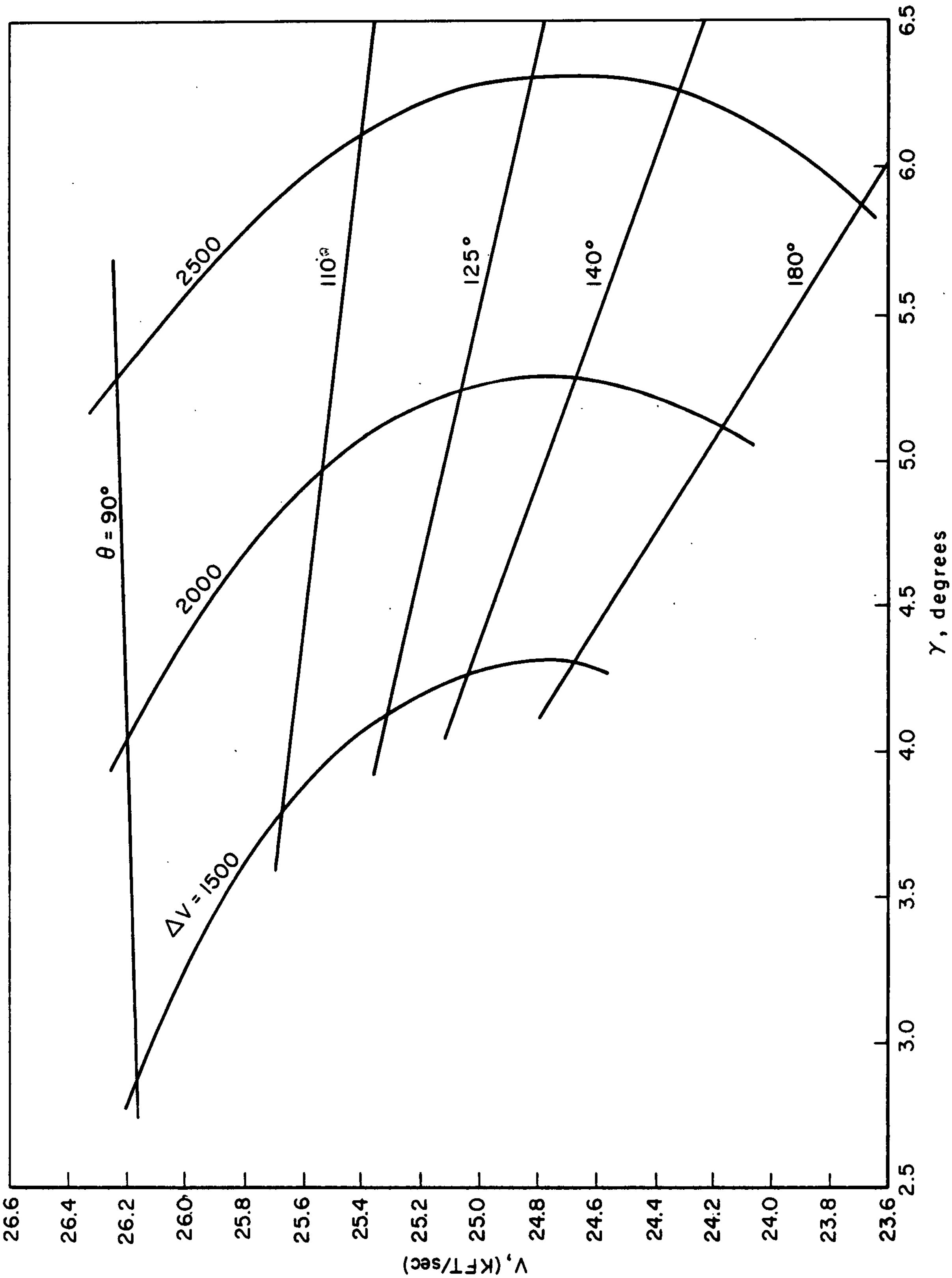


Fig. H-2—Conditions at altitude of 250,000 ft as a function of magnitude and angle of velocity increment
Circular orbit = 150 statute miles

increase to $\gamma = 4^{\circ}$ or 5°. Conditions at this altitude are shown in Fig. H-2 as functions of the impulse characteristics.

Radio observation of the vehicle immediately after the beginning of descent establishes a predicted vacuum path. This, together with estimated atmospheric effects, enables a prediction of an approximate impact area. The problem is similar to that described previously for the recovery of a scientific satellite through natural decay of the orbit.(2) In that case, the predicted impact area was a narrow strip, a few miles wide and several hundred miles long. In the present case, the steeper descent results in a smaller uncertainty in range error. Final recovery is accomplished by overflight search. This, of course, requires operation of the radio beacon after the water impact.

ATMOSPHERIC DRAG AND DECELERATION

The effect of the earth's atmosphere on the vehicle's path and velocity is dependent both on the approach path and on the vehicle's mass-drag characteristics. For even a rather shallow entry angle (say 5° or more), the path in the atmosphere is essentially linear - at least until after appreciable deceleration and heating have occurred. For this type of path, an approximate analysis(3) is sufficiently accurate for estimation of deceleration and heating. By this analysis, the velocity altitude variation is given by:

$$\frac{u}{u_1} = e^{-\frac{C_D A_c}{W \sin \gamma} \frac{\rho_{SL}}{2\alpha} \sigma} \qquad (1)$$

The velocity-altitude variation, as given by this equation, is shown in Fig. H-3 for several values of the drag-mass parameter, $C_D A_c / W \sin \theta$. It is seen that similar curves result, with relative displacements for various values of the drag-mass parameter. The maximum deceleration occurs when the velocity has been reduced to about 61 per cent of the initial value. The maximum deceleration is independent of the vehicle's size, shape, and mass and is dependent only on the initial velocity and on the entry angle. For the present case, the maximum deceleration amounts to about 20 g's. The altitude of maximum deceleration is dependent on the entry angle as well as on the drag-mass characteristics of the body.

As in any preliminary design, the tentative choice of parameters involves a cyclic process in which considerations of structure, heating, recovery, etc., all contribute to the final choice. Although the reasons for choosing various phases of the vehicle design will be developed in this Appendix, it is necessary for discussion purposes to describe that design here. The vehicle is essentially an ellipsoid with a maximum diameter of about 3 ft. While its orbital weight is 300 lb, about 50 lb (of propellant) is used to initiate descent from orbit, so that the weight at atmospheric entry is about 250 lb. This results in a drag-mass parameter, $C_D A_c / W \sin \gamma = 0.153$ sq ft/lb, which yields the velocity-altitude variation shown by the dotted line in Fig. H-4. A maximum deceleration of about 20 g's occurs at 110,000 ft altitude. Choice of the weight stemmed from booster capabilities and desired payload. Size and shape evolved from considerations of packaging and of heating and recovery. The film package, radio beacon, antenna, batteries, etc., are arranged to preserve entry stability and permit recovery.

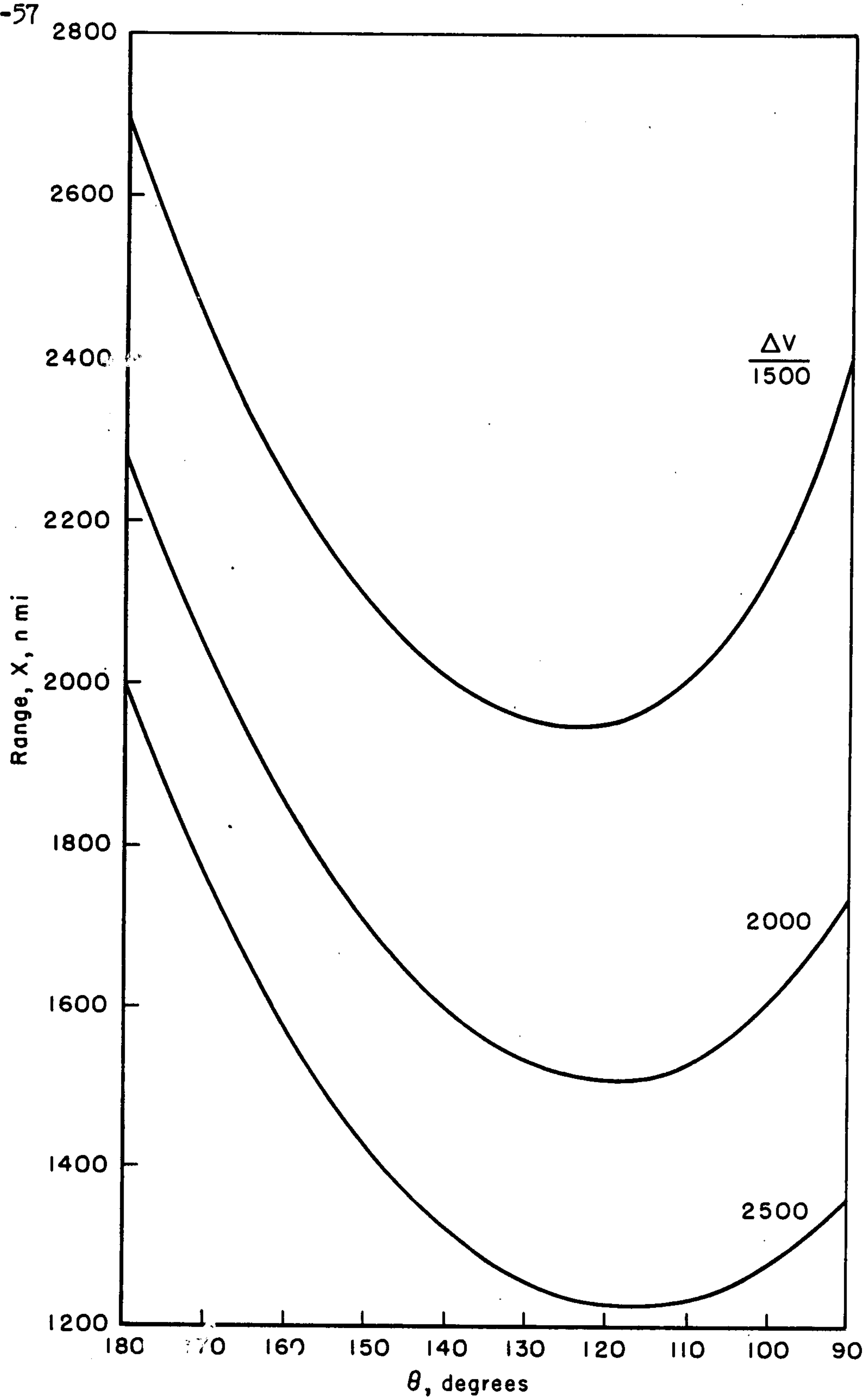


Fig. H-3—Vacuum descent range as a function of magnitude and angle of velocity increment

Circular orbit = 150 statute miles

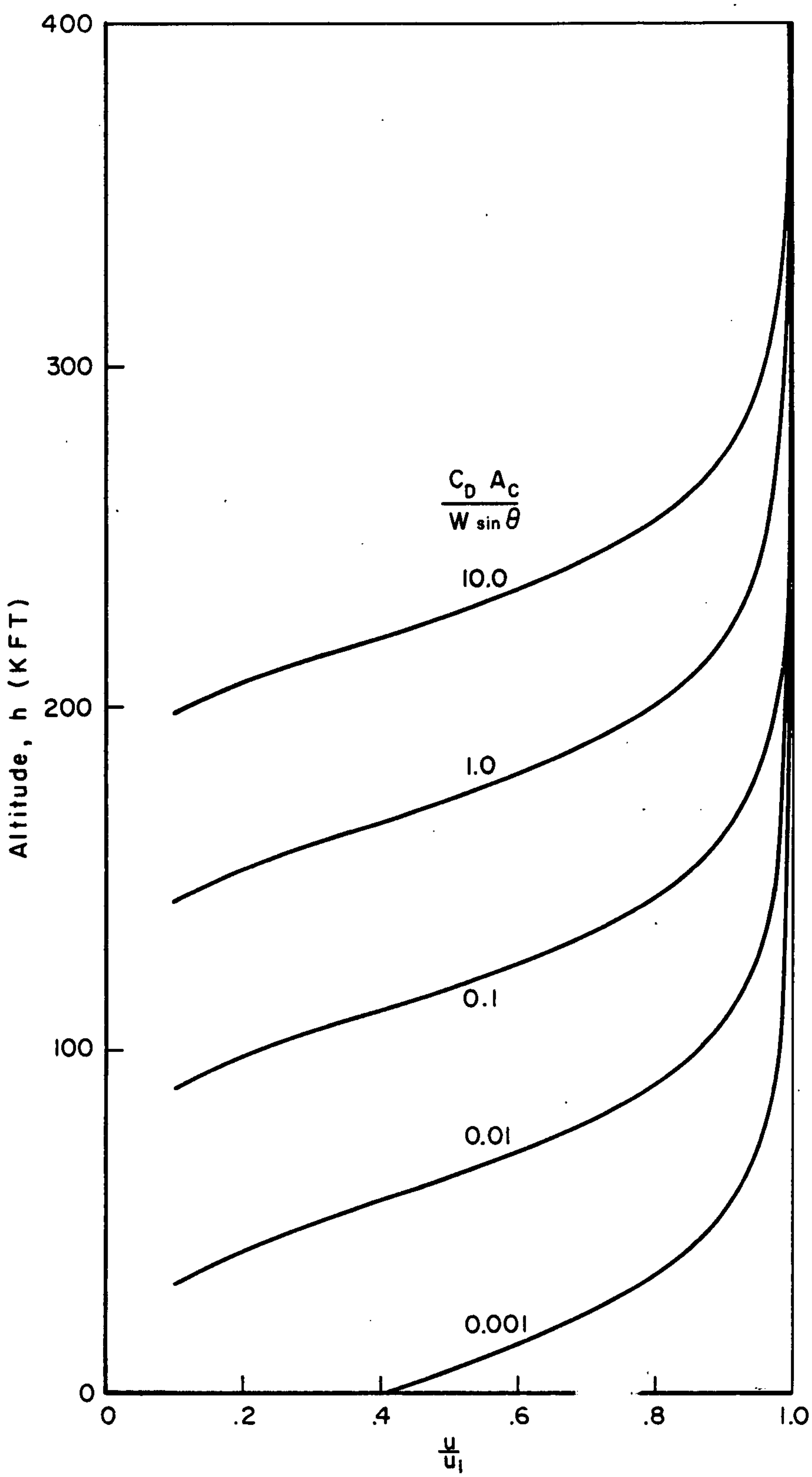


Fig.H-4—Velocity variation with altitude for atmospheric entry

As previously noted, the orientation of the vehicle is controlled by a moderate spin rate so that the vehicle is properly oriented for photography. The original orientation will be approximately preserved during the first part of the return trajectory, so that initial atmospheric entry will be rear-end first. By disp[illegible]ing the center of gravity toward the front of the body, aerodynamic forces can be made to re-orient the body. The design, of course, should be such that the re-orientation is completed before heating is appreciable.*

Ultimately the body falls vertically at terminal velocity:

$$u = \frac{1}{\sqrt{\frac{C_D A_c}{W} \frac{\rho_{SL}}{2g} \sigma}} \qquad (2)$$

which results in an impact velocity of about 400 ft/second for the body described.

RE-ENTRY HEATING AND SURFACE PROTECTION

During penetration of the atmosphere, a vehicle's kinetic energy is converted into thermal energy of the surrounding air. Some of this thermal energy is transferred to the body as heat. The rate of this transfer varies during descent both with air density and vehicle velocity. Heat is transferred by both convection and radiation from the hot gas 'cap' over the front of the body to the body's surface. The rates of both con-

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*A similar problem occurred in the case of the recovery of a circumlunar vehicle.(4) A dynamic analy[illegible] indicated that, for an essentially backward initial entry, the vehicle becomes righted at an altitude of about 250,000 ft. It then oscillates about the desired orientation; as the altitude decreases, the oscillations decrease in magnitude and increase in frequency. The predicted amplitude was about 10°, and the frequency about 15 cycles/second in the region of maximum heating and deceleration.

vective and radiative heat transfer increase with air density and vehicle velocity, and are thus most severe when high velocities are allowed to persist into the lower atmosphere.

Below about 300,000 ft altitude, the atmosphere is dense enough to give an effective continuum type of flow. Here a shock wave occurs ahead of the body and the thermal energy appears in the hot 'shocked' air between the shock wave and the body. Passage through the shock wave increases the air density by a factor of ten or so, increases the temperature ten- to fifty-fold, and causes appreciable dissociation and some ionization. Heat is transferred from this heated region to the vehicle surface by convection (and conduction) through the viscous boundary layer and by radiation from the hot gas. When the boundary layer is of the laminar type (say above about 100,000 ft altitude), the convective heating rate per unit frontal area may be approximated, for relatively blunt bodies, as[4]

$$\left(\frac{q}{A_c}\right)_{LC} = \frac{3}{\sqrt{\left(\frac{Re}{Md}\right)_{SL} d}} \frac{\rho_{SL}}{2} \sigma u^3 \qquad (3)$$

It will be noted that this indicates a variation in heating rate per unit surface area with the sine of the angle of surface inclination. A turbulent boundary-layer condition, occurring in the lower atmosphere, results in heating rates which are higher by about an order of magnitude.

Surface heating by radiation from the hot-gas region is still somewhat a matter of conjecture. Preliminary work, however, indicates that it becomes appreciable only when very high velocities are allowed to persist into the lower atmosphere. For the present case it is judged negligible.

The heating rate increases during the initial stages of entry because of the increasing atmospheric density. However, it reaches a maximum value when the velocity has been reduced to about 80 per cent of its initial value where the rate of decrease of velocity overcomes the rate of increase of atmospheric density. Thereafter, the heating decreases as additional deceleration occurs. As described more extensively elsewhere[3] the aerodynamic heat input can be balanced by thermal radiation from a thin metallic skin providing the heating rate is low enough so that the maximum equilibrium surface temperature is allowable for the surface material. However, this requires a rather light body or a drag-brake device so as to provide deceleration high in the atmosphere. In the present case, for the sake of simplicity, such complexity is not desired. Therefore, the radiation technique of heat rejection has been discarded in favor of some means of heat absorption.

One possibility is the use of a thick metallic skin to absorb the heat by temperature rise. Such a system is not very efficient from a weight standpoint, however, absorbing only about 100 Btu/lb.[5] More efficient is the use of a surface material which absorbs heat by a phase change. Heat absorptions of the order of 1,000 to 5,000 Btu/lb or more appear to be obtainable through the use of a material which vaprozes.[5][7] For example, it has been estimated that the depolymerization of Teflon will absorb somewhat more than 1,000 Btu/lb,[6] other plastic materials yield values up to 5,000 Btu/lb,[7] and graphite gives a value in the order of 10,000 Btu/lb. Advanced ICBM re-entry body designs involve the use of such ablation systems, and the Jupiter currently uses such a system. The required weight of such a vaporizing surface material depends on the total heat input during atmospheric penetration. If we integrate, for example, the laminar convective heating rate (Eq. (3)) over the time of entry, the total heat load is found to be:

$$\left(\frac{Q}{A}\right)_{LC} = \frac{3\sqrt{\pi}\,\rho_{SL} u_i^2 \,\mathrm{erf}\left(\sqrt{\frac{C_D A_c}{W \sin\gamma}\frac{\rho_{SL}}{\alpha}}\right)}{2\alpha \sin\gamma \sqrt{\frac{C_D A_c}{W \sin\gamma}\frac{\rho_{SL}}{\alpha}}\sqrt{\left(\frac{Re}{Md}\right)_{SL} d}} \tag{4}$$

and for a surface material having a heat absorption per unit weight, γ, the weight of cooling system is found to be, in ratio to the total weight,*

$$\frac{W_c}{W} = \frac{3\sqrt{\pi}\, u_i^2 \sqrt{\frac{C_D A_c}{W \sin\gamma}\frac{\rho_{SL}}{\alpha}}\,\mathrm{erf}\left(\sqrt{\frac{C_D A_c}{W \sin\gamma}\frac{\rho_{SL}}{\alpha}}\right)}{2C_D\lambda \sqrt{\left(\frac{Re}{Md}\right)_{SL} d}} \tag{5}$$

This equation yields the rather surprising conclusion that the cooling-system weight requirement increases with the drag-mass parameter. A somewhat similar expression can be derived for turbulent convective heating. A qualitative picture of the weight requirements for a cooling system or drag brakes is indicated in Fig. H-5 as a function of the drag-mass parameter. Two minima appear: one where the drag-mass parameter is slightly greater than that which would bring abeut heating in the lower atmosphere with consequent higher turbulent heating rates, the other where the drag-mass parameter is just large enough so that radiative heat transfer is possible and yet not large enough to require a large weight in drag brakes. To avoid the complexity of drag brakes, the first of these

*This, of course, presupposes that the ratio W_c/W is relatively small compared to unity, since the velocity variation, Eq. (1), assumes a constant mass. For larger values of the weight ratio, a meteor-type analysis which takes into account the changing mass would have to be used.

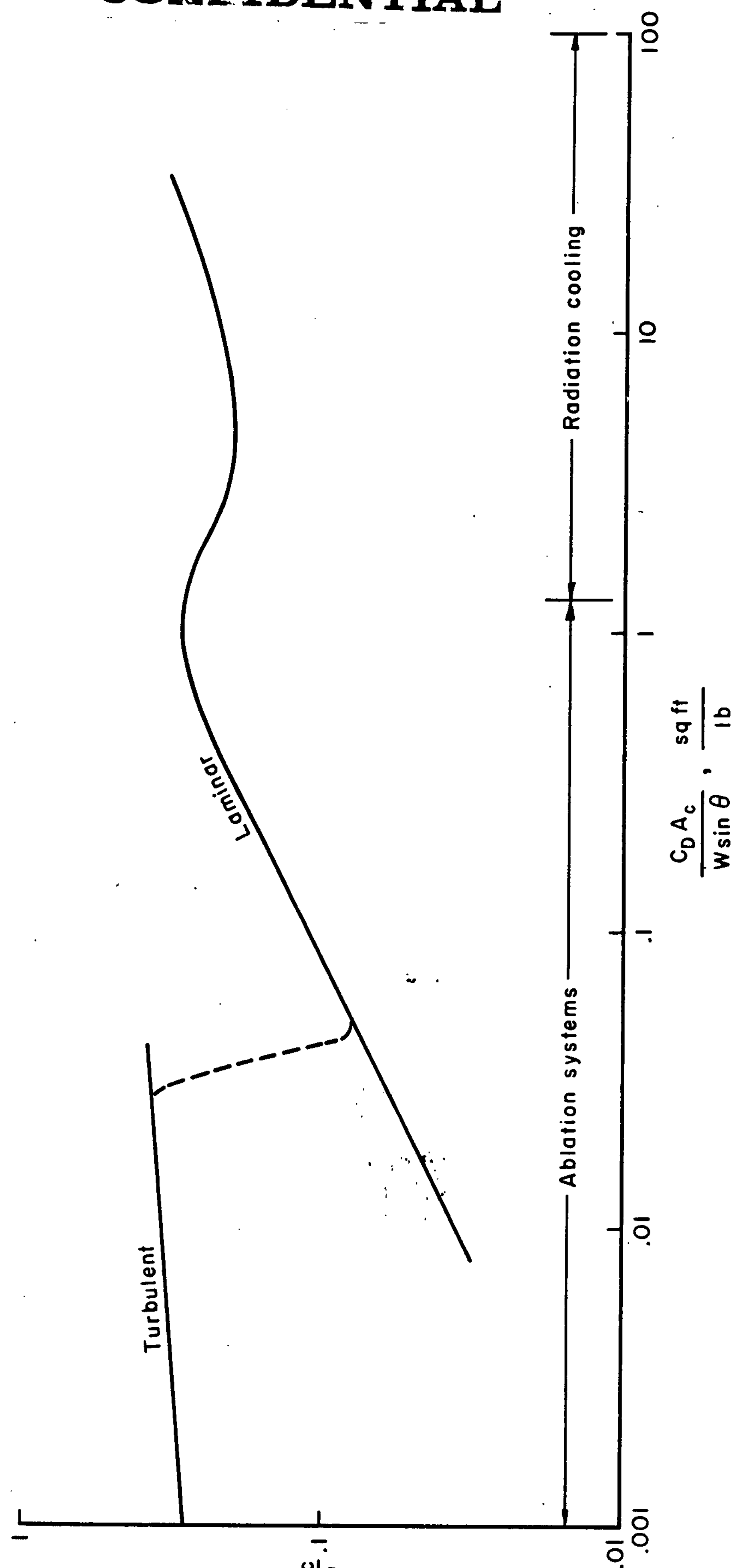

Fig. H-5—Cooling system weight requirements for recoverable satellite vehicle minima was chosen. The choice, as described previously, is an ellipsoidal body about 3 ft in diameter and weighing 250 lb at the beginning of re-entry.

Because of the shallow angle of atmospheric entry, the aerodynamic heating rates for this recovery are lower than those experienced by an ICBM re-entry body. However, the more gradual deceleration involves a longer time of heating so that the total heat load is greater, as Eq. (5) indicates. This means that a greater fraction of the total weight is required for heat protection. Thus, while only about 5 per cent of the total weight of an ICBM re-entry body may be ablated, the present case involves somewhat more than 10 per cent.

Using a vaporizing material with a heat-absorbing ability of 2,500 Btu/lb, about 35 lb of surface material is predicted to vaporize during descent. Using a relatively generous factor of safety, 65 lb of heat-absorbing surface has been postulated for this vehicle. This corresponds to an average thickness of about 3/8 in., with about one-half ablating and one-half remaining.

WATER IMPACT PHENOMENA[8]

In an initial program for the recovery of the satellite, it would be desirable to keep the return package as simple as possible. This would increase our ability to predict reliably the behavior of the vehicle during its re-entry. Therefore, a minimum of reliance should be placed on such devices as drag brakes, parachutes, reverse rockets and the like, just before impact. The satellite must survive impact on return to earth and the radio beacon must continue to operate after impact.

Analysis shows that the internal equipment can be protected and the deceleration loads kept to below 1,000 g's.

The film package, beacon and batteries are connected to the forebody by means of a properly chosen plastic, with properties compatible with the deceleration loads that can be tolerated by the internal components (several plastic materials with suitable combination of properties are available).

SYMBOLS FOR APPENDIX H

A_c frontal (cross-sectional) area of body

A_s surface area

C_D drag coefficient

d characteristic dimension of body

e base of natural logarithms

g gravitational acceleration

h altitude

M Mach number

q heating rate

Re Reynolds number, $du\rho/\mu$

u velocity

u_i velocity at initial entry condition

W mass of body

x distance

α exponential coefficient in density relation, $\sigma = e^{-\alpha h}$

ρ atmospheric density

ρ_{SL} atmospheric density at sea level

σ density ratio, $\rho\ \rho_{SL}$

γ path angle with local horizontal

μ viscosity

Subscripts

LC laminar connection

REFERENCES TO APPENDIX H

1. Huntzicker, J. H., H. A. Lieske, Physical Recovery of Satellite Payloads - A Preliminary Investigation (U), The RAND Corporation, Research Memorandum RM-1811, June 26, 1956 (Secret).

2. Gazley, Jr., C., D. J. Masson, A Recoverable Scientific Satellite, The RAND Corporation, Paper P-958, February 27, 1957.

3. Gazley, Jr., C., Deceleration and Heating of Bodies Entering a Planetary Atmosphere from Space, The RAND Corporation, Paper P-955, February 18, 1957.

4. Gazley, Jr., C., D. J. Masson, Recovery of a Circum-Lunar Instrument Carrier, The RAND Corporation, Paper P-1119, August 19, 1957.

5. Masson, D. J., C. Gazley, Jr., 'Surface-Protection and Cooling Systems for High-Speed Flight,' Aeron. Engr. Review, Vol. 15, No. 11, November 1956, pp. 46-55.

6. Porter, R. W., Recovery of Data in Physical Form, Franklin Institute Monograph No. 2, June 1956, pp. 103-112.

7. Masson, D. J., Mass-Transfer Cooling for Hypersonic Flight (U), The RAND Corporation, Classified Paper S-51, June 24, 1957 (Secret).

8. Masson, D. J., The RAND Corporation, Personal Communication, October 1957.

Appendix J

ON THE INFORMATION DELIVERED

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There are several different ways to measure the information content of a picture, and hence to assess the worth and rate of delivery of reconnaissance information from a satellite. From the standpoint of information theory and communication engineering, it is customary to calculate the number of bits involved and to evaluate the amount and rate on this basis. However, every discernible leaf on a tree and patch of cloud is counted as information on this basis, and the photo-interpreter finds this a strange measure indeed. He prefers to evaluate photography in terms of useful detail he can see in the picture and the extent of the usable coverage. These are subjective judgments which depend not only on the picture but also on the use to which it is put, and may be poorly correlated with the number of bits. Between these extremes, and perhaps serving as a bridge between them, is the assessment in terms of scale, resolution, and area covered. In this appendix the proposed system is approximately evaluated from each of these three viewpoints.

From the information theory and communication viewpoint, the information content of a picture is equal to

$$2581\ N^2 A \log_2 G \quad \text{bits}$$

where

N = photographic resolution in lines/mm

A = area of the picture in square inches

G = number of distinct levels of grey in the photograph.

(The factor 2581 consists of a factor of 4 to convert photographic lines to electronic scanning lines and a factor to reconcile the mixed-up dimensional units in N and A.)

This expression tacitly assumes complete equivalence of a photograph and its TV replica when each photographic 'line' is covered by two electronic 'lines'. Unfortunately, it is not at all clear that such a replica is indeed equivalent. At least in part this results from the uncertainty with which photographic resolution is related to recognition of pictorial details; it is well-known that a photo-interpreter often can see detail which he 'shouldn't' be able to see. Consequently, the expression above is best regarded as a lower limit to the number of bits contained in a picture. A better guess might be about 10 times this expression.

Insofar as photographs at different scales, resolutions, and contrasts are equally well assessed by the expression above, it provides a means to compare the information-collecting capabilities of various photographic systems. However, for most practical purposes this expression is needlessly complicated by the factor $\log_2 G$. Consider the following table which spans the range of probable interest.

Number of distinct greys	$\log_2 G$	Relative information content
6	2.58	0.661
8	3.00	0.768
10	3.32	0.850
12	3.58	0.918
15	3.91	1.000
20	4.32	1.106
30	4.91	1.256
40	5.32	1.362

The last column of the table has been centered arbitrarily on G = 15 in order to point out that the information content of a picture is fairly insensitive to the number of grey levels. If it is assumed that all pictures exhibit about 15 levels of grey the estimate arrived at is not likely to be wrong by more than 30 per cent. In view of the other uncertainties involved in estimating the information content, this is not a serious error. Hence the original expression can be replaced by

$$10{,}000\ N^2A \text{ bits.}$$

Using the foregoing fairly reasonable, but somewhat arbitrary, approach it is possible to tabulate a relative comparison of photographic systems; three of present interest are shown below.

System	Area A	Resolution N	No. of bits
One conventional 9 x 9 aerial photo	81	10	8.1×10^7
One day's take of original Lockheed satellite; 15 feet of 2-1/4 inch film	405	100	4.05×10^{10}
One mission of recoverable system proposed here; 500 feet of 4-1/2 inch film	27,000	40	4.32×10^{11}

This analysis indicates that the original Lockheed satellite would deliver each day the information-content equivalent of approximately 500 conventional 9 x 9 aerial photographs. Further, the recoverable system proposed herein would deliver in one day (one mission) somewhat more than 10 times the one-day take of the original Lockheed satellite, or about 5300 ordinary 9 x 9 photographs.

The information-theory approach is a questionable measure of the worth of all this information, because it ignores the undeniable influence of such factors as scale on the practical utility of the photographs. However, this approach is fundamental to an analysis of the problems of electronic transmission of photographs by TV, wirephoto, or similar techniques. A transmission channel (radio or wire) having an effective bandwidth B cycles per second can deliver at most

$$B \log_2 (1 + S/N)$$

bits per second. Here S/N is the ratio of signal power to noise power at the receiver. This bit-rate is a theoretical upper limit and cannot be surpassed by any system. However, real systems rarely even come close to this limit. For favorable values of S/N (perhaps 10 db or better) the theoretical value can be approached only if very elaborate coding schemes are used. Such elaboration leads to high cost, high complexity, low reliability, and undue sensitivity to changes in S/N. Consequently, such an approach is rarely used, and would be inappropriate in a satellite system. To assess a practical situation it is more appropriate to demand a signal-to-noise ratio better than 10 db and then to assume an information rate of B bits per second (something between 1/3 and 1/10 of the theoretical limit, depending upon the S/N which is achieved.).

Combining the two expressions above yields a practical lower limit to the time which must be consumed in transmitting a picture:

$$\frac{10{,}000\ N^2 A}{B} \quad \text{seconds}$$

For the original Lockheed satellite, using $B = 6 \times 10^6$ (which is close to the maximum practical value at present) the time required to transmit one day's take turns out to be 112-1/2 minutes.

It is not practical, because of line-of-sight limitations, to transmit to a ground station for longer than about 10 minutes on any one pass of a 300-mile satellite. Such a satellite makes about 16 passes per day. Consequently, the daily take of the original Lockheed satellite pretty well fills the communication capacity to a single receiving station, even if the satellite passed over that station every time. The characteristics of the orbit and the rotation of the earth prevent the satellite from passing over a single station on every pass, and so several stations are needed to receive the 112-1/2 minutes per day of transmission.

The early recoverable satellite proposed here will achieve about 10 times the daily take discussed above. To match this a telemetering satellite using only one 6 megacycle channel would need 18 3/4 hours of transmission. Such a schedule would be out of the question because receivers would be needed on the oceans and in unfriendly territory. The growth potential of the recoverable satellite is such that later models may well deliver 20 times as much information per day. A comparable daily rate by telemetering would require some 200 to 500 megacycles of bandwidth and a long-time average power radiation of more than 300 watts (not including any power consumption aboard the satellite). Such numbers are fantastic in terms of the present state of the art. They point up emphatically the intrinsic virtue of physical recovery of photographs. This conclusion is quite independent of detailed considerations of focal length, format, resolution, etc., and applies as fully to the immediate future as it does to system growth.

The foregoing discussion of information delivery is certainly relevant to the military geographer and mapper, but it is not couched in the quantities to which he is accustomed. The area covered by the proposed 500 feet of 4 1/2 inch film is 4 million square miles. The scale at the center of the photograph is 750,000. Neglecting earth curvature, the average scale at the far point of the sweep, which is about 300 miles wide for a nominal design altitude of about 140-150 miles, is approximately 1,300,000. (This average scale is the geometric mean of the scales in the x and y directions on the photograph.)

The photo interpreter is not primarily interested in either of the assessments above; he is concerned with what he can see in the picture. This is much more difficult to estimate, first because there is so little relevant evidence based on similar small-scale photography, and, second, because this estimate is so dependent upon subjective judgments. The photo interpreter is not accustomed to looking at pictures on Plus-X Aerecon film at a resolution of 40 lines/mm; he is used to looking at Aero Super XX at 10 lines/mm.

A rough comparison of the proposed photography with presently available photography is furnished by considering 10-lines/mm photography at scale numbers of about 200,000. This is precisely what can be obtained on oblique metregon photography from the Air Force standard tri-met charting installation at altitudes of 40,000-50,000 feet. This scale number occurs out near the principal point of such photography, about 50° to 60° off vertical.

Examination of such photography, and of the available rocket photography that is applicable to this question, leads to the firm belief that the proposed camera installation, operating at a ground resolution of about 60 feet at the center of photography, will indeed detect most, if not all, uncamouflaged airfields, transportation lines, canals, rail lines, industrial centers, and the like. Port areas, marshalling yards, and defense missile sites similar to those in the Moscow area should be easily identified and located. If previous cover is available, new construction such as might be characteristic of large missile sites, should be detected.

Appendix K

ON THE GROWTH POTENTIAL OF THE RECOVERABLE FILM RECONNAISSANCE SATELLITE

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Clearly the major emphasis of this proposal is on the first and the easiest recoverable photographic reconnaissance satellite. This version, the 12-inch f/3.5 camera using 500 ft of 5-inch film, and based on the Thor booster, is seen as the first of a series of such cameras and systems. This system is capable of doing reconnaissance at Level A in adequate detail. As one learns how the system works, gains confidence in satellite operations, and understands the environmental constraints and the intelligence problems, it becomes possible to consider more advanced systems. This early model would be followed by a 36-inch focal-length camera system, using 1500 ft of 9-inch film. It is believed at this time that this larger camera can also be put on orbit using a Thor-type booster system, with a maximum payload-stage weight of about 300 lb.

This second reconnaissance system could begin to do job B in the four levels of reconnaissance with adequate precision and detail. For the really long term, considerable growth potential may be anticipated for the panoramic type of camera system, with the eventual use of a 10-ft focal-length system employing about 2500 ft of 18-inch film; however, this camera would require not the Thor but the Atlas booster. The time phasing of the several projects could be approximately as follows: availability of 12-inch systems, one year from date of contract; availability of 36-inch system, 18 months; availability of 10-foot system, 3 years from the start of work. This information is summarized graphically on p. 21.

It is important to recognize that this type of system will have a usefulness into the indefinite future, even after the availability of satellite reconnaissance systems which are able to talk back to the ground. The reason for this lies in the potential usefulness of extremely long focal-length, recoverable reconnaissance systems. For reasons discussed in Appendix J, it is unlikely that the capacity and detail-gathering capability of a surveillance satellite, limited as it is by bandwidth hours to talk back to the ground, will ever approach the data gathering rate which seems to be available within a few years in the long focal-length recoverable system.

Table 1 compares the camera performance of the Lockheed 117L 6-in. and 36-in. systems with that of the three systems discussed in this report.

It was not the purpose of this report to develop an argument either for or against 'talk-back' satellites vis a vis recoverable satellites. Clearly there is a role for both kinds. Some jobs will be done better by one, some by the other. They should be complementary, not competitive.

Table 1

SATELLITE CAMERA PERFORMANCE COMPARISON

Performance	LMSD I	LMSD II	RAND PAN 1	RAND PAN 3	RAND PAN 10
Focal Length	6 in.	36 in.	12 in.	36 in.	120 in.
Altitude	300 mi	300 mi	142 mi	142 mi	142 mi
Resolution (lines/mm)	100	100	40	40	40
Film	microfile	microfile	Aerecon	Aerecon	Aerecon*
Strip Width s.mi.	100	16	300	300	100
Lens Speed	f/3.5	f/3.5	f/3.5	f/3.5	f/3.5
Shutter Speed	$\frac{1}{50}$ sec	$\frac{1}{100}$ sec	$\frac{1}{4000}$ sec	$\frac{1}{4000}$ sec	$\frac{1}{4000}$ sec
Center Scale	3,000,000	500,000	750,000	250,000	75,000
Average Edge Scale	3,000,000	500,000	1,300,000	440,000	82,000
Ground Resolution Feet Per Line at Center	100	16	60	20	6
Ground Resolution Feet Per Line - Average at Edge of Format	100	16	105	35	6-1 2
Total Ground Area Covered (sq. mi)	9,000,000 10 days	750,000 30 days	4,000,000 500 ft x 5 in. film 1 day	2,900,000 1500 ft x 9-1/2 in. film 1 day	750,000 2500 ft x 18-1/2 in. film 1 day
Total Film Area Sq Ft	25	75	185	1125	3750

*The film here called Aerecon is perhaps more properly titled Plux-X-Aerecon.

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